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PRELIMINARY DESIGN STUDY FOR A HI-HICAT VEHICLE AND INSTRUMENTATION SYSTEM

FOX CONNER

LOCKHEED-CALIFORNIA COMPANY

TECHNICAL REPORT No. AFFDL-TR-66-100

OCTOBER 1966

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200 - January 1967 - C0192-19-384

PRELIMINARY DESIGN STUDY
FOR A
HI-HICAT VEHICLE AND INSTRUMENTATION SYSTEM

FOX CONNER

FOREWORD

This document is the final report of a study of a vehicle and instrumentation system for measuring High-High Altitude Critical Atmospheric Turbulence (HI-HICAT). The investigations were conducted by the Lockheed-California Company at Burbank, California, under Contract AF33(615)-2569. The contract was performed under Project No. 1469, "Vehicle Loads Validation", Task No. 146902, "Critical Atmospheric Turbulence". The study was monitored by J. N. Garrison, FDTE, of the Air Force Flight Dynamics Laboratory at Wright-Patterson Air Force Base, Ohio.

The study was conducted from 1 June to 31 December 1965 and the report was submitted 28 February 1966.

This study is indebted to assistance from outside organizations. Their experience in certain fields led to a better understanding of the feasibility of various concepts and permitted a further detailing of the HI-HICAT vehicle and instrumentation system. Special mention is accorded the following organizations for their time and effort in preparing proposals which analyzed and recommended configurations for the subsystems listed below:

Recovery System	Northrop Ventura, 1515 Rancho Conejo Boulevard, Newbury Park, California 91320
Propellant and Propulsion System	Rocketdyne, 6633 Canoga Avenue, Canoga Park, California 91304
Q-ball	Northrop Nortronics, 1 Research Park, Palos Verdes Peninsula, California 90274

This manuscript was released by the author in February 1966 for publication as a RTD Technical Report.

This technical report has been reviewed and is approved.

James C. Horsley Jr.
JAMES C. HORSLEY, JR.
Major, USAF
Chief, Experimental Mechanics Branch

ABSTRACT

A preliminary design study was conducted on an unmanned HI-HICAT (High-High Altitude Critical Atmospheric Turbulence) vehicle and instrumentation system to measure turbulence at altitudes from 70,000 to 200,000 feet. The vehicle configuration selected as optimum for this extreme range of altitudes is a parawing. For the study, emphasis was placed on designing a system for the middle portion of the altitude band from 100,000 to 150,000 feet. In this band a lifting body configuration is competitive with the parawing. Both systems feature a one-stage vehicle which is air launched from an F-4C aircraft at supersonic speeds. A cluster of eight P4-1 rocket chambers accelerates the vehicle up to cruise speed. The vehicle cruises in horizontal flight at speeds as high as Mach 6 until propellant exhaustion or until the sustainer engine is shut down. It then decelerates at the cruise altitude to obtain additional data miles. Recovery is initiated when the vehicle slows down to Mach 1.5. An air snatch completes the mission. Turbulence data is gathered by a digital system and stored on a magnetic tape recorder and telemetered back to the launch aircraft, the recovery aircraft, and any available ground station. An inertial navigator supplies attitude and acceleration data, but the fine scale attitude motions in turbulence are obtained from a package of three precision rate gyros. A one-axis, servoed Q-ball is recommended as the flow direction sensor. The total RDT&E, production, and operating cost of 500 data gathering flights is estimated at $53 million.

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SECTION 1

INTRODUCTION

Clear air turbulence has been a serious design problem at all altitudes at which airplanes have flown. Figure 1 (Reference 1) indicates that turbulence is a major factor in determining the fatigue life of an aircraft. It is expected that this contribution may increase for future flight vehicles due to the combined effects of increased speed, altitude, size and flexibility.

To date, turbulence data, applicable to aircraft design, has been gathered at altitudes as high as 70,000 feet. This data has been gathered only recently and by aircraft already in service and thus has not been available as an input into the basic aircraft design. The result has been the loss of life and vehicles, and expensive design "fixes". With this hindsight it becomes possible to foresee the necessity of obtaining turbulence design criteria for advanced vehicles before they reach the final design phase. Many of these vehicles will operate at altitudes above 70,000 feet, for which there is presently no data available. The HI-HICAT (High-High Altitude Critical Atmos. Turbulence) program was initiated to collect turbulence data between 70,000 and 200,000 feet altitude.

Thus far turbulence information in this high altitude regime has been limited to vertical profiles of wind velocities, and has been obtained from rising rockets or falling instruments ejected from rockets. Descriptions and understanding of the turbulence fields at these levels, sufficient for use in aircraft design criteria, requires extensive and accurate measurements in the horizontal, rather than the vertical.

This preliminary design study was initiated with the objective of evaluating a vehicle and instrumentation system which could operate in horizontal flight at altitudes from 70,000 to 200,000 feet. Specific requirements as set forth in the contract are that the system must be capable of obtaining in-flight data sufficient to permit definition of HI-HICAT with respect to location, extent and intensity, and associated meteorological parameters. The data must be sufficient to permit analysis of the turbulence by power spectral density methods. Emphasis will be placed on measuring long wavelength turbulence (up to 75,000 feet per cycle) by operating the vehicle in horizontal flight through areas of high wind shears, jet streams, gravity waves and similar phenomena. For an adequate data sample, the cruise range of the vehicle should be approximately 25 times the maximum wavelength of interest. For a wavelength of 75,000 feet, a range of 350 miles is required (statute mile used herein). The primary requirement on the instrumentation is that it must be capable of resolving a minimum rms (root-mean-square) gust velocity of 1 fps. The number of flights to be considered for study purposes is 50 minimum and 1000 maximum.

Since the primary objective of the HI-HICAT program is to measure turbulence, it is necessary to keep in mind the fact that all other systems must be designed around the instrumentation system. The vehicle must be designed with the philosophy that its purpose is to carry the instrumentation package with minimum influence on instrumentation design or function. Thus, throughout this study, the vehicle and instrumentation have been considered as a single entity.

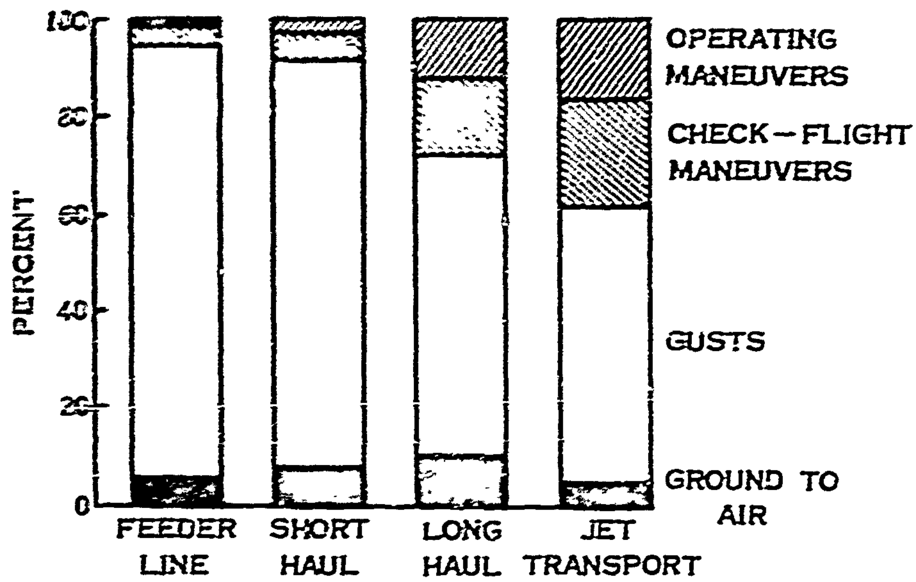

FIGURE 1. FATIGUE DAMAGE ATTRIBUTED TO VARIOUS LOAD SOURCES

This study thus consisted of selecting an instrumentation system to perform the above stated mission, followed by vehicle configuration studies, and integrating these into an overall operational system including optimum launch method, propulsion, guidance and control, and data retrieval.

This study is part of an overall program to statistically define the characteristics of high altitude, clear air turbulence and to verify or correct existing theories on the power spectral density of turbulence at these altitudes. The establishment of a turbulence model, to be used as a basis for predicting the fatigue loading of aerospace vehicle structures, will result from the collected data of the overall program.

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SECTION 2

SUMMARY

One-stage, unmanned, parawing and lifting body designs were evolved for a HI-HICAT vehicle and instrumentation system capable of measuring turbulence at altitudes from 70,000 to 200,000 feet. The mission for either configuration consists of the same series of flight phases, as illustrated in Figure 2. This study concludes that the optimum vehicle would be air launched. This would be accomplished as follows: An F-4C aircraft enters the launch maneuver at near Mach 2 at 35,000 feet and pulls up until the HI-HICAT vehicle is released at the pre-set launch angle. The helium pressurization system is activated and the rocket engines fire to boost the HI-HICAT vehicle up to speed. The boost engines (but not the sustainer engine) are shut down and the vehicle "coasts" up to the cruise altitude, at which point the parawing deploys, if installed. The sustainer engine is modulated to provide the required level of thrust in cruising flight. The parawing vehicle cruises at speeds near Mach 3 at 70,000 feet and Mach 6 at 200,000 feet. At the lower altitudes the cruise speed can be set for maximum range, but at the higher altitudes the cruise speed for maximum range is too high and must be limited due to material temperatures. After propellant exhaustion or after the engine is commanded to shut down, the vehicle decelerates at the cruise altitude to Mach 1.5, the recovery initiation speed, or to a maximum angle of attack of 50 degrees. The deceleration adds valuable data miles. If the maximum angle of attack is reached first, the vehicle glides down until the speed drops to Mach 1.5. After the main parachute opens, an air snatch aircraft picks up the vehicle and returns it to the operating base for refurbishing.

Although the external configurations are radically different, the internal systems for either the parawing or the lifting body design are the same except for one major variation. The parawing vehicle has a pressure-fed engine system, whereas the lifting body has a turbopump-fed system. All available space must be filled with propellants, and complex shapes result for a lifting body design which are inappropriate for a pressure-fed system. The installation of a turbopump requires a more complex design, but the freedom in choosing an aerodynamic configuration compensates for this disadvantage. Admittedly, only a parawing can achieve satisfactory range at extremely high altitudes, but the lack of aerodynamic and thermodynamic test data gives a degree of uncertainty about a parawing design which can only be resolved by a systematic parawing research program. Lifting body designs have been subjected to considerable research, development, and test effort; therefore, a HI-HICAT development program could proceed immediately without a research phase.

The recommended, unmanned vehicle configuration is dependent on altitude. For the overall 70,000 to 200,000 feet band, the parawing is recommended. It is the only vehicle capable of flying at 200,000 feet in a practical manner. For flight at lower altitudes, smaller wings, or no wings, can be attached to the same basic fuselage. In the mid-altitudes (100,000 to 150,000 feet), rigid shapes are competitive and the lifting body vehicle is recommended.

At the lower altitudes of interest, the YF-12A aircraft will have application. Past turbulence investigations have always employed a manned aircraft. The highest altitudes explored to date have been with a U-2 aircraft on the HICAT program. An instrumented YF-12A aircraft is the next logical step. The cost of obtaining data, although greater than for past investigations, would be less than for the unmanned systems studied herein.

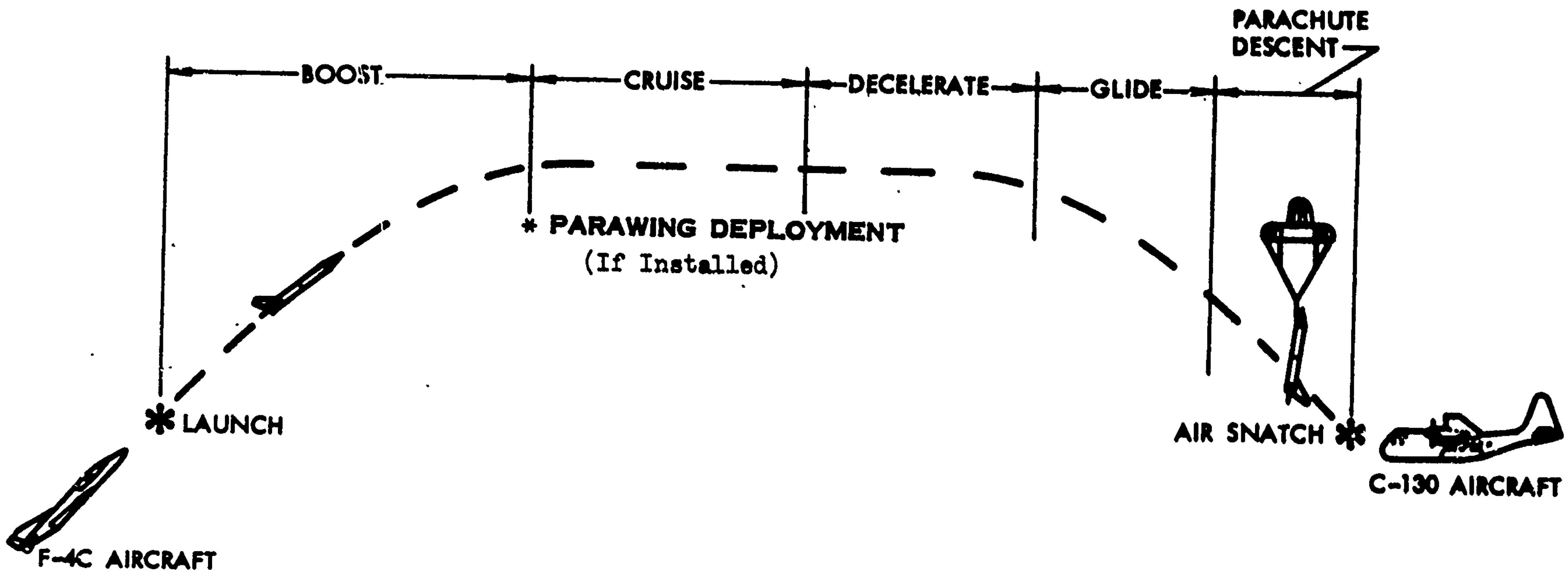

FIGURE 2. MISSION PROFILE

A flow direction sensor is provided to measure the fine scale lateral and vertical variations in the winds relative to the HI-HICAT vehicle. The response of the vehicle, in turn, to long scale turbulence wavelengths, is measured by accelerometers, and rate and position gyros. The recommended flow direction sensor is a one axis, servoed Q-ball design similar to that developed and in operation on the X-15 research aircraft. The high supersonic speeds of the HI-HICAT vehicle requires that extremely small variations in the angle of attack and sideslip be measured. The accuracy attainable with the overall system depends primarily on the accuracy of the flow direction measurements. The Q-ball design described herein promises accuracies of 0.005 deg. at frequencies from 1/20 to 7 cps. The overall system accuracy is 0.01 deg. for the same range of frequencies. At the instrumentation design Mach number of 4 at 70,000 feet and Mach 6 at 200,000 feet, the system is capable of measuring an 0.8 fps rms gust at wavelengths from 400 to 78,000 feet at the lower speed and a 1.0 fps rms gust at wavelengths from 600 to 94,000 feet at the higher speed and altitude.

The instrumentation system is designed to sample 20, 13-bit words simultaneously at a rate of 40 per second. Two of the 20 channels are required for the angle of attack since the measuring range will be greater than the capability of a single channel. Both on-board recording and telemetering of data are considered in the design, and both functions can be provided on a single flight, if desired. The magnetic tape recorder has a capacity greater than 13 million bits, sufficient for 15-minute flights. The telemetering includes a 5-watt transmitter and a slot or flush cavity antenna.

A total system cost of 27, 53 and 80 million dollars is estimated for either the parawing or the lifting body vehicle for 50, 500 and 1000 flights, respectively. These figures include developing, testing, engineering, production and operating costs, but do not include the cost of research on the parawing. Since the technology for supersonic parawings is poorly developed, a 20-month additional research program must be completed before a go-ahead can be authorized for a HI-HICAT parawing system, at a cost of two million dollars.

Three views of the parawing and the lifting body systems are presented in Figures 3 and 4. The major design parameters and features are described in Table 1.

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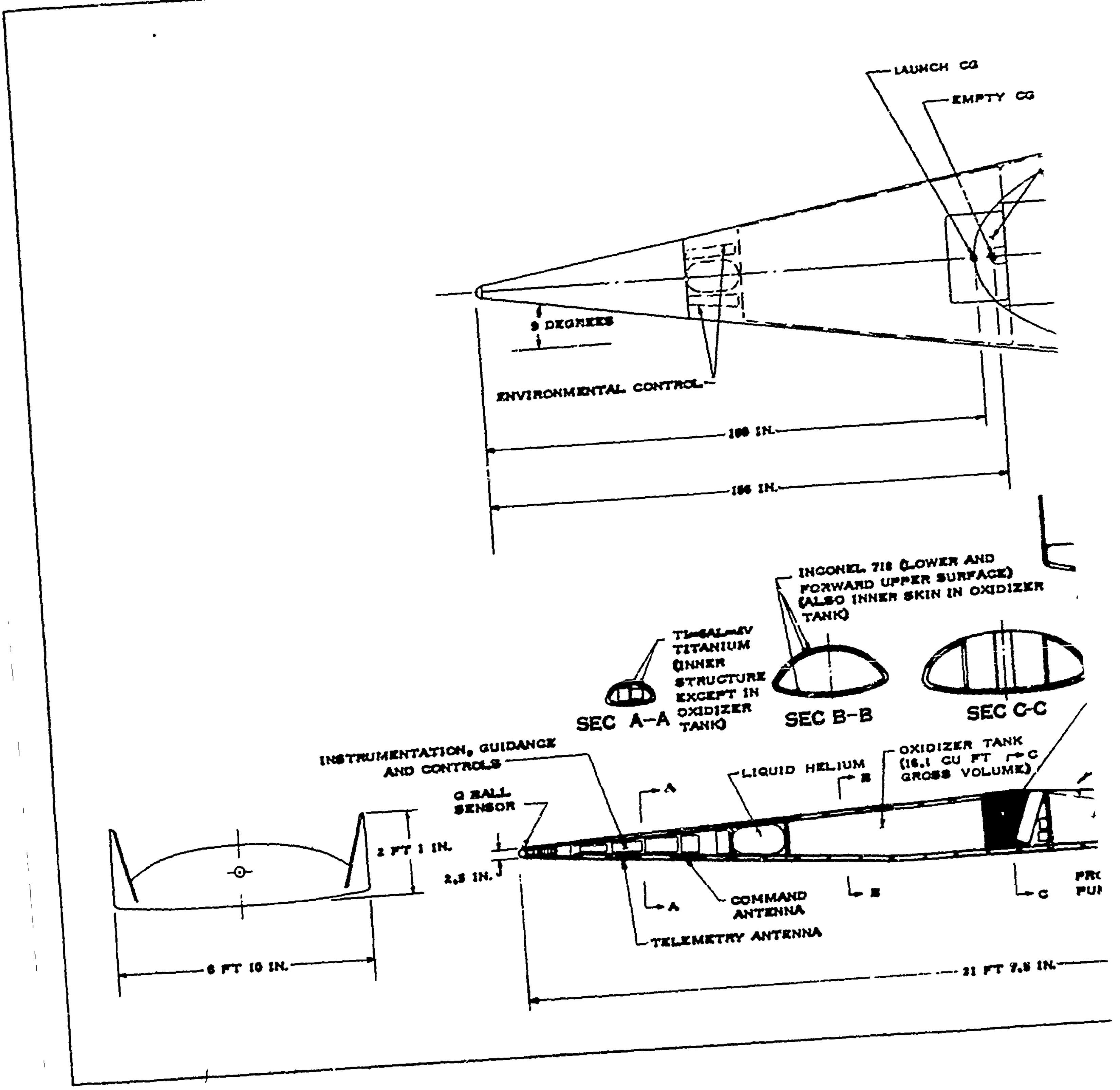

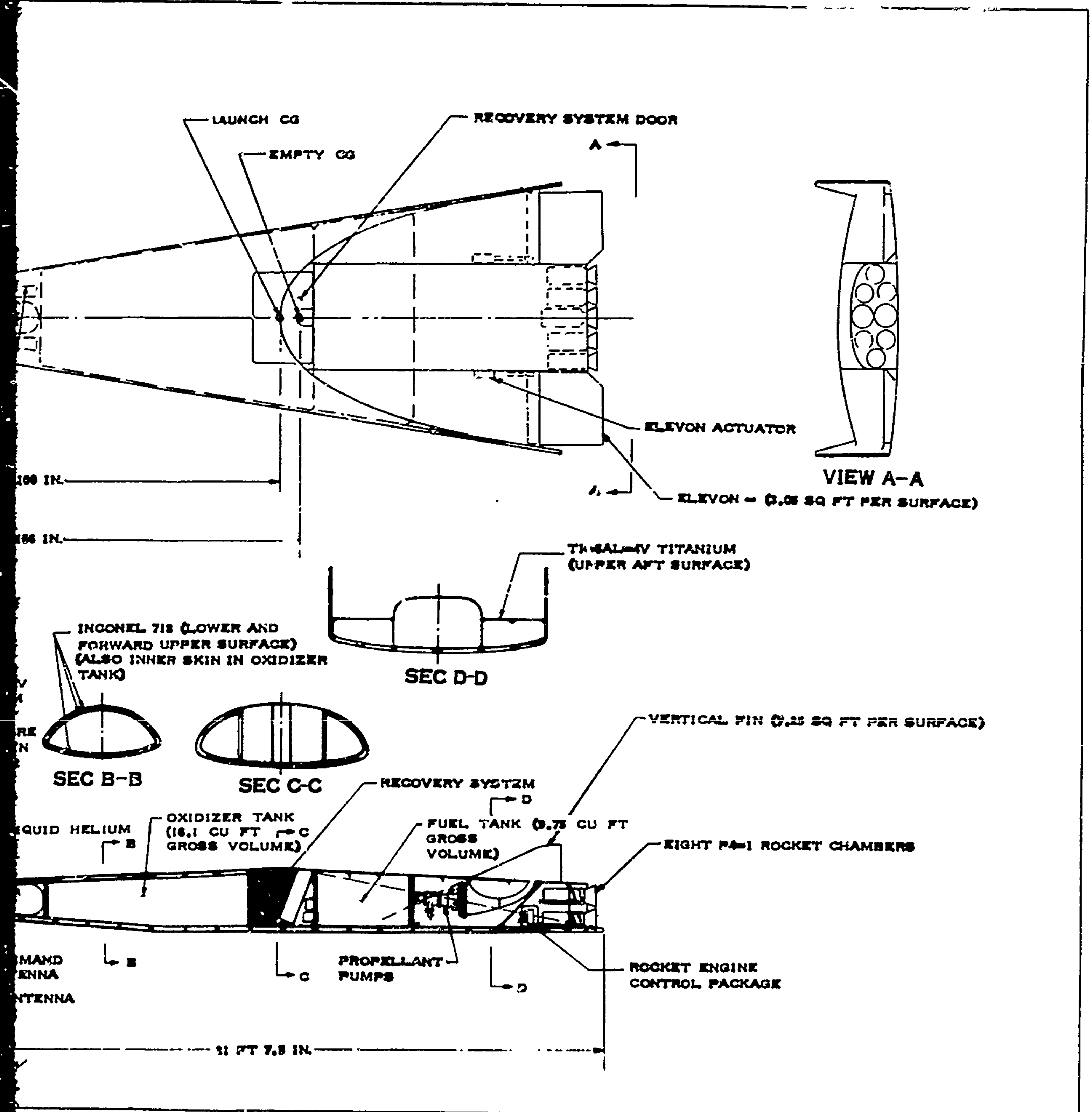

FIGURE 3. LIFTING BODY VEHICLE, THREE VIEW

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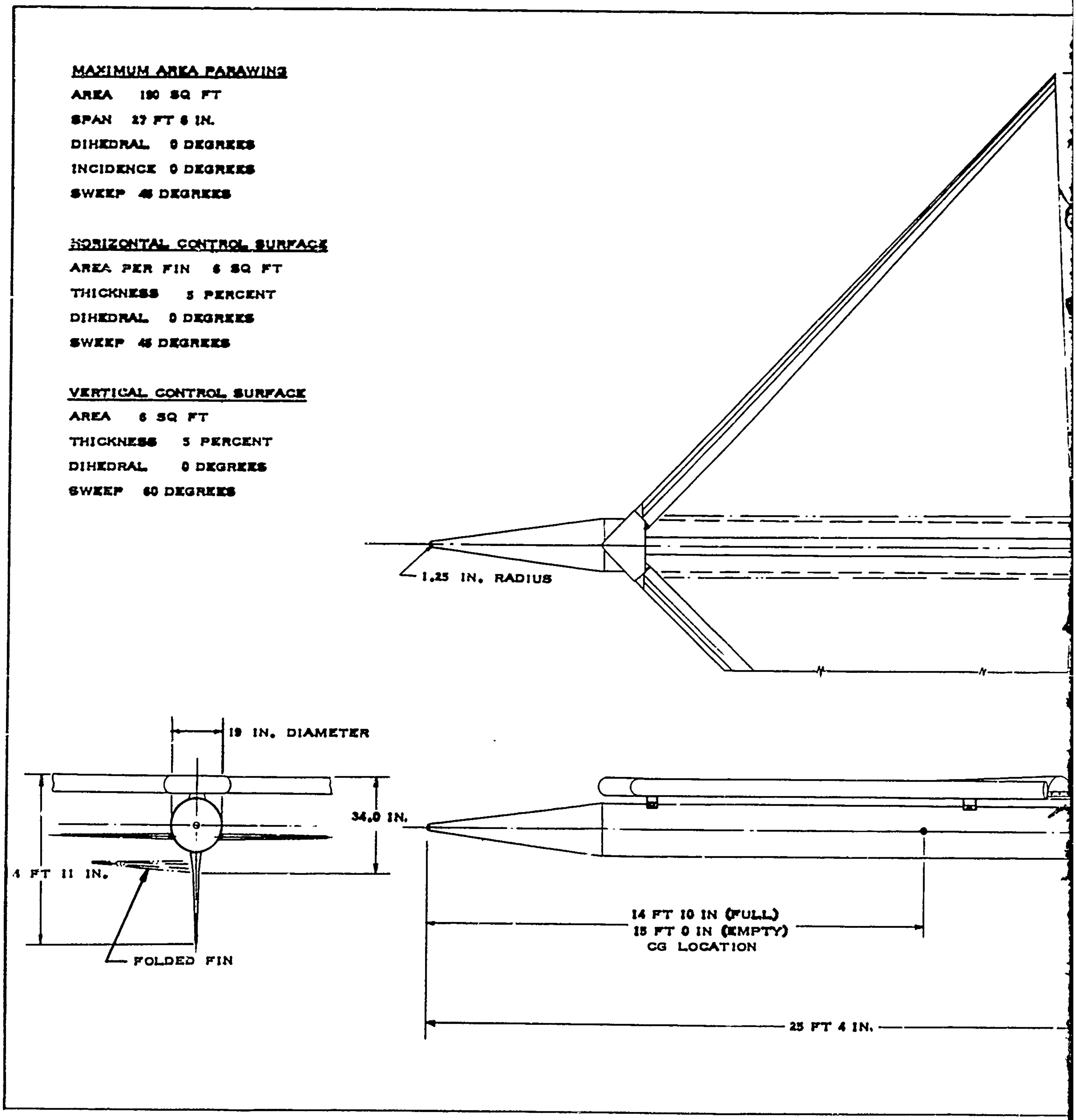


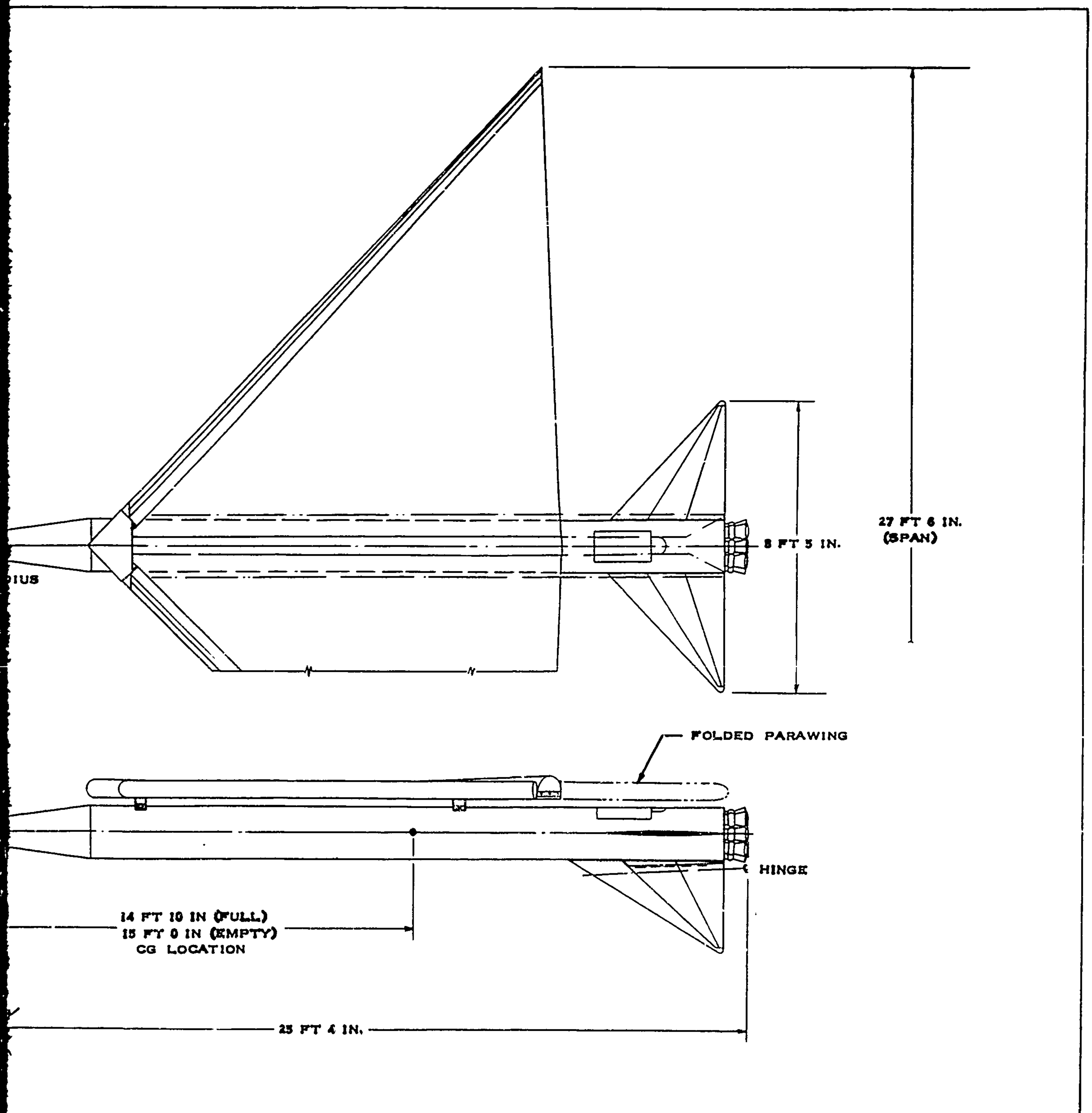


FIGURE 4. PARAWING VEHICLE THREE-VIEW

TABLE 1

SUMMARY OF SYSTEM DESIGN FEATURES

	Lifting Body	Large Parawing
Weight		
Full	3262 lb	2873 Lb
Empty	1157 lb	1036 Lb
Airframe		
Length	21 ft 7.5 in.	25 ft 4 in.
Span	6 ft 10 in.	27 ft 6 in.
Height	2 ft 1 in.	4 ft 11 in.
Planform area	78.5 sq ft	190 sq ft
Leading edge sweep	81 deg	45 deg
Leading edge diameter	3 in.	4 in. (minimum)
Q-ball diameter	2.5 in.	2.5 in.
Primary structural material	Ti-6Al-4V titanium and Inconel Alloy 718	Ti-6Al-4V titanium and Inconel Alloy 718

	Either System
Propellant and Propulsion System	
Type	Eight clustered P4-1 liquid rocket chambers
Fuel	Hydyne MAF 4
Oxidizer	IRFNA
Mixture ratio (oxidizer-to-fuel weight ratio)	3.0:1
Expansion area ratio	
Boost engine	14.0:1
Sustainer engine	23.5:1
Maximum thrust	5660 lb (lifting body - 150,000 ft) 4935 lb (parawing - 200,000 ft)
Pressurization and Cooling System	
Type	Cyrogenic helium
Helium tank pressure	1000 psia
Recovery System	
Type	Two stage parachute followed by air snatch

TABLE 1 (Concluded)

	Either System
Command and Control System	
Type	Inertial navigator with "ground" control override
Controls	Aerodynamic surfaces, engine throttle, engine shutoff, and recovery initiation
Instrumentation	
Type	PCM (digital)
Flow direction sensor	One-axis servoed Q-ball
Data handling	On-board magnetic tape recording and telemetering
Data channels	20
Samples per second per channel	40
Capacity	Greater than 13×10^6 bits

SECTION 3

CONFIGURATION EVALUATION

This section is concerned with the identification and comparative evaluation of the possible configuration candidates for carrying the instrumentation package through its mission. Also included in this section is the selection of the various supporting systems for propulsion, launch and recovery.

3.1 MISSION PROFILE DEFINITION

Fundamental to selecting a vehicle configuration is the definition of the performance requirements. The HI-HICAT vehicle must be capable of cruising horizontally for a distance of 350 miles at all altitudes between 70,000 and 200,000 feet. The complete profile may be broken down into the following phases (See Figure 2): (1) launch, (2) boost, (3) cruise, (4) deceleration, (5) glide and (6) recovery. The configuration of the HI-HICAT vehicle will be directly affected by the approach selected for the first three of these mission phases and only indirectly affected by the other flight phases. The discussion in this section, therefore, will cover the first three phases only since only a preliminary defining of the HI-HICAT mission is intended.

For purposes of completing the definition of the HI-HICAT vehicle mission profile, a preliminary estimate of the cruise velocities was made. Figure 5 is a plot of dynamic pressure ($q = \frac{W/S}{C_L}$ where q is the dynamic pressure, W/S the wing loading and C_L the lift coefficient). At the highest altitudes the highest lift coefficients will be required and a maximum for design would be expected to be roughly 0.5 for supersonic cruise. It may be seen from Figure 5 that high supersonic speeds are a must in spite of the thermodynamic problems that will be encountered. For example, even at Mach 6 at 200,000 feet a wing loading of only 5 will be required at a C_L of 0.5, an unusually low value. (It will be shown later that speeds much greater than Mach 6 result in excessive material temperatures.)

At low altitudes, low speeds are favored to reduce the variation in the range of cruising C_L's with a consequent improvement in the lift-to-drag ratios. Subsonic speeds are unlikely, however, because the longitudinal trim requirements are simplified by the constancy of the aerodynamic moments at supersonic speeds. Also, any gain in lift-to-drag ratio may be offset by the greater fuel consumption, per mile, of a rocket. (The reasons for choosing a rocket engine are discussed in Section 3.1.3.) And finally, the flight time is shortened at supersonic speed by a number of minutes with a consequent reduction in the size of secondary power sources.

3.1.1 Launch

Two launch methods were considered, ground launch and air launch. Air launch is recommended. A major disadvantage of ground launch for the large number of flights to be considered (up to 1000) is the necessity of using a booster for each launch. The booster cannot be made recoverable and reusable in any practical manner. Thus, the cost of a new booster would be incurred for each flight. Analyses of booster costs indicate that these would probably be nearly 50 percent of the total operational cost of each

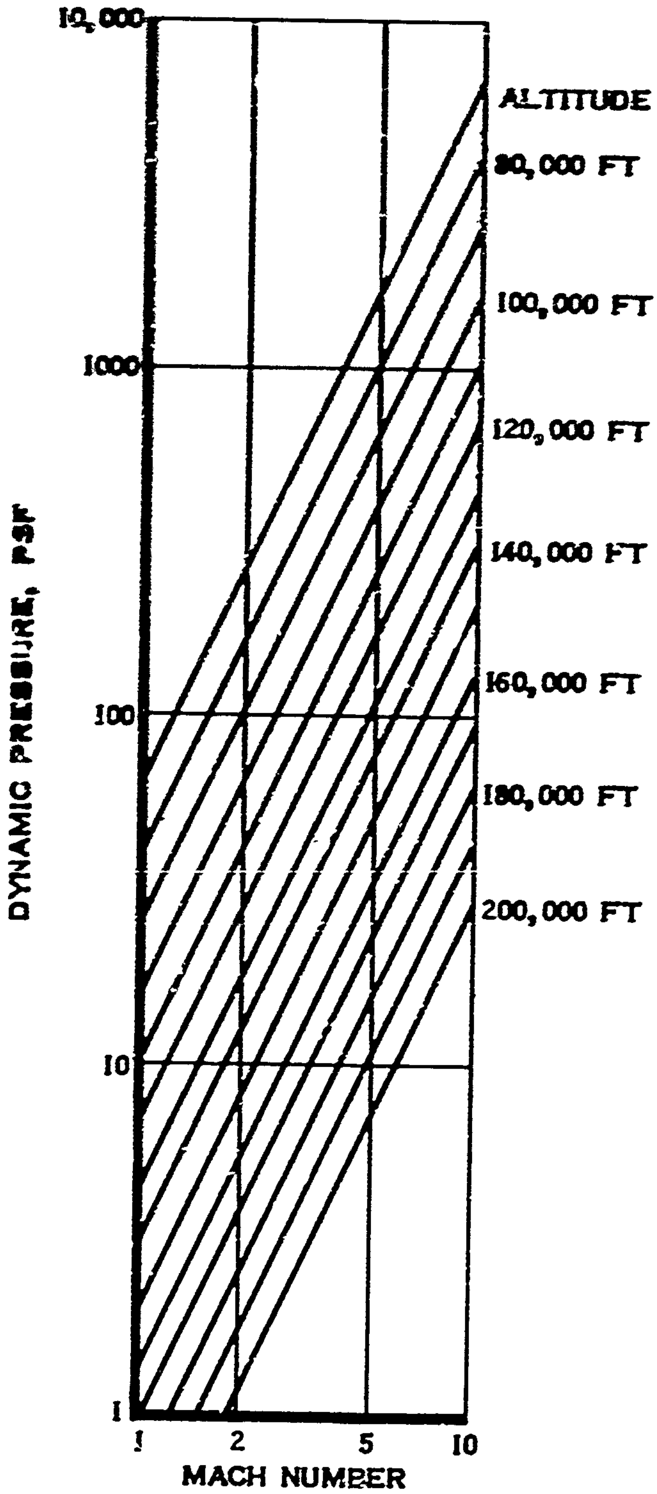


FIGURE 5. DYNAMIC PRESSURE

flight. Another major problem associated with ground launch is the high accelerations and high dynamic pressures placed on the vehicle and instrumentation during the low altitude boost portion of the mission.

By launching the HI-HICAT vehicle from an aircraft at high altitude, the above stated problems can be greatly reduced or eliminated. Launch costs are much less since the launch aircraft is reusable. The loads imposed on the HI-HICAT system during aircraft climb-to-altitude are also much less than those for ground launch. Other advantages of air launch over ground launch are:

1. Reduction in the velocity increment required for boost.
2. Reduction in drag losses during boost.
3. Mobile, worldwide launching platform.
4. High reliability of an aircraft compared to a booster.
5. Fewer launch site restrictions.

An air launch is not without its disadvantages. Typically:

1. The vehicle is subject to buffeting and vibration before launch.
2. Tests must be conducted to verify that the launch aircraft and HI-HICAT vehicle combination possess satisfactory flight qualities, performance and launch characteristics.

Experience with other programs indicates the advantages of an air launch outweigh the disadvantages if a large number of launches are contemplated. Typical of past and present programs with repeating air launches are the X-15 research aircraft, the X-7 ramjet research vehicle and the AQM-37A target drone.

3.1.2 Boost

Two HI-HICAT vehicle boost/cruise configurations were studied; a one-stage system and a two-stage system. In the one-stage system the booster is an integral part of the cruise vehicle. The booster is reusable and is recovered with the cruise vehicle. In the two-stage system the booster is ejected from the cruise vehicle prior to cruise flight. The booster can then either be recovered or expended.

The expendable booster suffers from many of the same handicaps as those previously enumerated for a ground launch booster and is not considered to be practical. Figure 6 presents a plot of the bare motor costs for solid propellant rockets. The band shown covers the range from a low production, conservative estimate to a high production, optimistic estimate. The curve is based on unsophisticated motors and excludes the costs of other expendable items such as tail fins or interstage assemblies. For a two-stage rocket-powered cruise vehicle weighing from 1000 to 1500 pounds cruising at speeds from Mach 4 to 6, a total impulse of roughly 500,000 lb-sec is required. It is evident that booster costs would be between $9000 and $25,000 per launch just for the bare motor alone. This cost is excessive for a program in which as many as 1000 missions are contemplated.

The recoverable booster must be reusable in order to justify recovering it. This implies that it uses liquid rockets. Reusable solid rockets have not been sufficiently developed to consider them for the HI-HICAT program. This

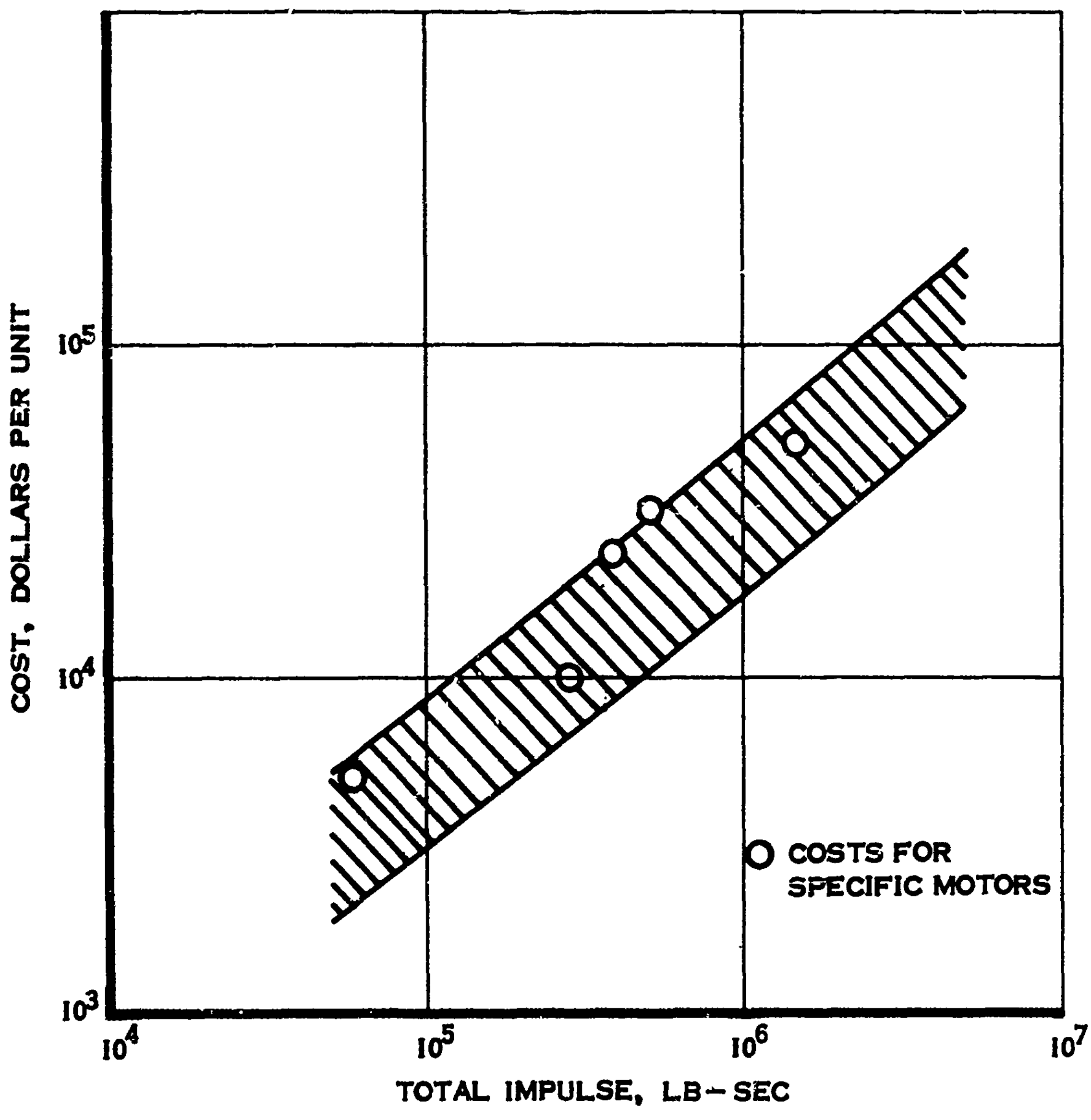


FIGURE 6. BARE MOTOR COSTS FOR SOLID PROPELLANT ROCKETS

fact also applies to the one-stage booster rockets. Thus both the one-stage and two-stage systems would use liquid propulsion.

The primary advantage of the two-stage system in which the booster is released prior to cruise, is the reduced weight to be carried during cruise. This in turn reduces the requirements on the vehicle design and propulsion system, resulting in, perhaps, lower one-time and recurring costs than those of a one-stage system. However, counteracting this possible advantage is the added cost and complexity of providing a separate recovery system for the booster. The liquid propulsion one-stage boost-cruise system is thus recommended on the basis of simplicity and reliability.

3.1.3 Cruise

A liquid rocket engine system for the cruise vehicle is included in the recommended HI-HICAT system. The justification for such a system is presented below by considering the arguments for rejecting alternate systems.

3.1.3.1 No Cruising Engine: Ballistic Vehicles and Dynamic Gliders

It is possible to conceive of systems which do not employ propulsion for cruising. Two such possibilities are described below:

1. A ballistic vehicle could make measurements as it went over the top of its trajectory. A range of 50 miles could be obtained with an altitude variation of less than ±5000 feet for an apogean speed of 5320 fps.
2. A concept of greater complexity is a glider which would be boosted up to the desired altitude. At that point, the guidance and control system would pitch up the vehicle to a lifting attitude. The vehicle then would become a dynamic glider and be controlled to decelerate at altitude until the maximum usable lift was obtained. The deceleration range would be of the order of 100 miles for the same apogean speed given above for the ballistic vehicle.

These concepts are best compared to a two-stage cruising vehicle with the same type of booster unit for the first stage. For such a vehicle the range would be greater than 350 miles and the weight would be roughly 1400 pounds. The weights of the pure ballistic vehicle and the dynamic glider are estimated at 200 to 400 pounds, respectively. Hence, all three vehicles have a weight-to-range ratio of roughly four. The cost per data mile would be less, therefore, for the larger vehicle because the cost of boosting a pound goes down as the vehicle weight goes up. Admittedly, the lack of a wing, a cruise powerplant, a fuel system and a complex guidance and control system causes the purely ballistic vehicle to look attractive. No savings can be effected for instrumentation, however, a major cost element. Consideration of the need for a large number of data miles for turbulence statistics leads to the conclusion that a cruising vehicle will result in a lower cost per data mile. Another factor is that all the vehicles discussed here are two-stage vehicles. The arguments in favor of discarding all types of two-stage vehicles were presented in Section 3.1.2

3.1.3.2 Air-breathing Powerplants

A ramjet could greatly increase the range of a HI-HICAT vehicle, as compared to a rocket, over a band of altitudes centering around the design altitude. Although a reduction in operating costs is probable, the cost of developing a new ramjet design solely for the HI-HICAT vehicle is excessive. In

comparison, a rocket does not have an operating ceiling and is the only engine which is feasible at the highest altitudes of interest.

A word should be said about air augmentation for a rocket engine, which could raise the specific impulse 30 to 50 percent for practical levels of air scooping (Reference 2). It remains to be demonstrated that the theoretical gains can be achieved in a practical flying system. Also, a disadvantage exists in that different size scoops would be required for different altitudes. Air augmentation is best considered for a possible future improvement of the basic HI-HICAT vehicle after the initial development is completed.

3.1.3.3 Solid Rockets

Very slow burning solid rockets have not been widely developed. Add to this the desire for reuse and throttling and it is apparent the cruise motor should not be a solid rocket.

3.1.3.4 Liquid Rockets

The recommended propulsion for the cruise vehicle is a liquid rocket. This is the only practical system for meeting the requirements of long range (350 miles) flight at nearly constant altitude. As will be shown in Section 3.3, a propulsion system can be built up with relative ease from the technology of a system already in existence, the LR64-NA-4 liquid rocket engine.

3.2 EXTERNAL CONFIGURATION ANALYSIS

There are a large number of lifting configurations which can be conceived for the HI-HICAT vehicle. They fall under two broad categories: rigid and expandable. Expandable configurations are of interest since a very large wing area is desirable for flight at 200,000 feet. The maximum dynamic pressure during boost will be two orders of magnitude greater than the dynamic pressure during cruise at 200,000 feet. Such a large wing cannot be made structurally strong enough to survive the dynamic pressures which will be encountered during boost. Somehow the wing must be retracted or folded into a compact shape for boosting. However, rigid shapes are not to be neglected. Moderately low wing loading will exist for shapes such as the lifting body and these configurations are usually easier to protect from the thermal environment.

Only aerodynamic controls will be considered for any type of lifting configuration. If a configuration is capable of generating enough lift to support itself, then it is reasonable to expect that aerodynamic controls can be designed which will achieve satisfactory pitching, rolling or yawing moments without recourse to reaction controls. Reaction controls are not really feasible for the HI-HICAT vehicle. The necessity for maintaining trim would put a steady drain on the propellants for the control jet and the propellant weights would become totally unrealistic. A variation of the reaction control concept is to swivel the rocket engine but since a portion of the mission is with power off, this variation must also be rejected.

3.2.1 Expanding Wings

Many concepts have been suggested in the literature but the only one which has been tested at supersonic speeds to any extent is the parawing. These concepts include inflatable shapes with an upper and lower surface: shapes which are pressurized by a gas or filled with a quick setting foam. Research is at a very early stage on these ideas and this study will not attempt to establish which concepts are practical or even feasible.

Also within the class of expandable structures are the mechanically folding wings. Unfortunately, a wing of very large area will require a number of hinge lines or hinge axes in order to fold within a roughly cylindrical shape for boosting. Such structures are not practical for this application.

The parawing, then, appears to be the most practical expanding structure for flight at high supersonic or hypersonic speeds and high altitudes. The typical parawing consists of a two-lobe sail, a keel, two leading edge booms and two spreader bars. The keel, the booms and the spreader bars can be inflatable columns, but such a structure is not desirable for the HI-HICAT mission. The diameter of inflatable leading edge booms must be large in order to withstand bending loads, but large diameters result in a high drag configuration. With rigid booms, diameters of a few inches are practical which is about the size desirable from a thermodynamic standpoint. A parawing design with rigid leading edge booms is studied in detail herein and will be shown to be of reasonable weight.

It is desirable, in the interest of structural simplicity, to avoid a separate tail for control. To achieve roll control, a pair of actuators could function as controllable spreader bars by differentially varying the sweep of the leading edge booms. Longitudinal control is more difficult and uncertain, unfortunately. Reference 3 considers the possibility of installing cables which pass through a tube along the trailing edge of the sail. These cables could be reeled in or out to vary the camber of the parawing and hence generate a varying pitching moment. Such a type of control is a distinct possibility, but this unusual concept requires research. In the interest of providing an aerodynamic control system which requires no research, a separate tail is provided with both a horizontal and a vertical surface. Such a tail can provide independent control about all three axes and allows the guidance system the greatest possible flexibility.

Rapid control movements are possible by utilizing tail surfaces of a delta planform which have a relatively fixed center of pressure and by rotating the entire control surface about a line that passes close to the center of pressure. Low hinge moments result and small actuators will respond rapidly enough to control the dynamic motions of the HI-HICAT vehicle.

The fuselage of the parawing vehicle studied herein, is a cylinder with a conical nose for structural simplicity. A nose fineness ratio of about four to one was chosen. A longer nose could be used which would reduce the drag at zero lift slightly, but if the nose is much longer some of the propellant would have to be stored in the nose in a tapered tank, an undesirable structure, since tank walls of varying thickness are required for minimum weight.

One vertical tail mounted on the underside of the fuselage is used in the parawing vehicle studied herein. T-tails would be structurally difficult for the thermal environment to be encountered and only a slight reduction, if any, in drag would be expected at high supersonic speeds. The vertical tail must be located on the underside of the fuselage instead of the upper since at hypersonic speeds a dynamic pressure "shadow" exists above the fuselage at high angles of attack.

Drawings of the proposed parawing configuration are presented in Figures 4, 7, and 8.

3.2.2 Fixed Wings

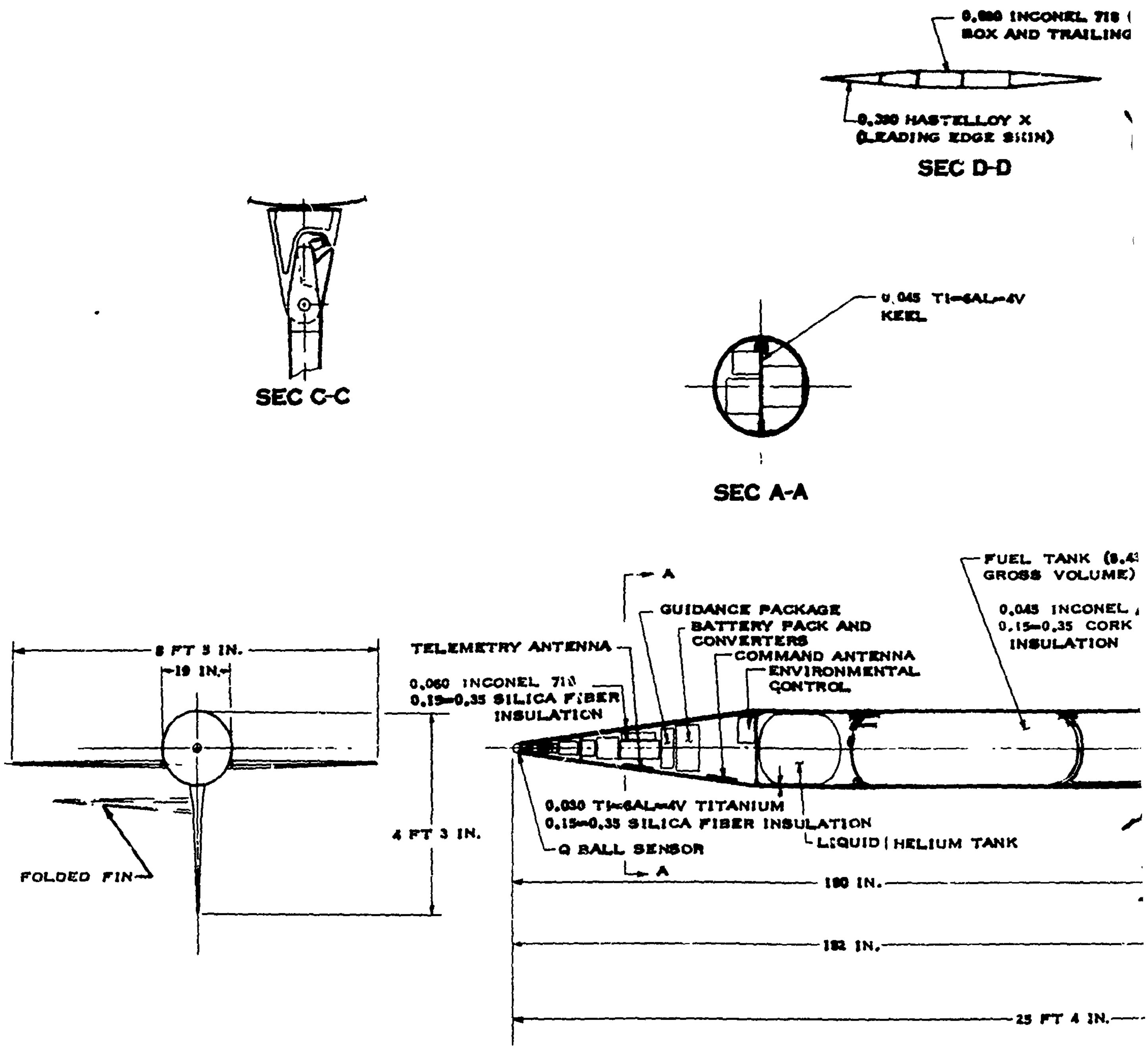
A separate tail does not appear to be needed for a fixed configuration. Many lifting bodies, planar bodies, etc., as well as existing supersonic designs, have done well without a horizontal tail and with only one or two vertical fins.

The next consideration to shape the HI-HICAT vehicle is the observation that vehicles designed for high supersonic speeds characteristically have a wing of high sweep. Low sweep designs are conceivable, but the wing must be so thin that it cannot contain an appreciable amount of propellants. For a HI-HICAT vehicle striving for maximum range, designs featuring large internal volumes are desirable and for this, lifting body designs are eminently suitable. Lifting body designs are being pursued vigorously and shapes having more favorable lift-to-drag ratios are expected to be generated. A development program leading to the production of HI-HICAT systems would be expected to allow for a complete search of the latest available data and to also allow for a moderate amount of wind tunnel tests on related shapes. Such a program should result in a shape of near optimum efficiency. The shape studied in detail herein is presented in Figure 3.

3.2.3 Existing Vehicles

Two vehicles already exist which would be feasible at the lower altitudes of interest. Unmanned designs were evolved in this study because manned designs are exceedingly complex and such vehicles will not be available in the near future for flight at the higher HI-HICAT altitudes. However, a prime candidate is the YF-12A aircraft at the lower HI-HICAT altitudes. It is capable of ranges many times greater than that for unmanned configurations of reasonable size. Much of the instrumentation used for other turbulence programs might be adaptable. Obviously, it could generate data at much lower costs at its operating altitudes. Unfortunately, security prevents a detailing of the performance of this aircraft. Also, there is a question as to the availability of this aircraft for turbulence research.

The other vehicle is the AQM-37A drone which is capable of speed and altitude beyond Mach 3 and 100,000 feet. Its use at the lower altitudes is discouraged by a range considerably less than the desired 350 miles, by a small payload capability, and by the superiority of the YF-12A vehicle for the HI-HICAT mission.



0.080 INCONEL 718 (
BOX AND TRAILING
0.390 HASTELLOY X
(LEADING EDGE SKIN)
SEC D-D
SEC C-C
0.045 TI-6AL-4V
KEEL
SEC A-A
FUEL TANK (9.4
GROSS VOLUME)
0.045 INCONEL
0.15-0.35 CORK
INSULATION
A
GUIDANCE PACKAGE
BATTERY PACK AND
CONVERTERS
COMMAND ANTENNA
ENVIRONMENTAL
CONTROL
TELEMETRY ANTENNA
0.060 INCONEL 718
0.15-0.35 SILICA FIBER
INSULATION
8 FT 3 IN.
19 IN.
4 FT 3 IN.
0.030 TI-6AL-4V TITANIUM
0.15-0.35 SILICA FIBER INSULATION
LIQUID HELIUM TANK
Q BALL SENSOR
FOLDED FIN
A
190 IN.
182 IN.
25 FT 4 IN.

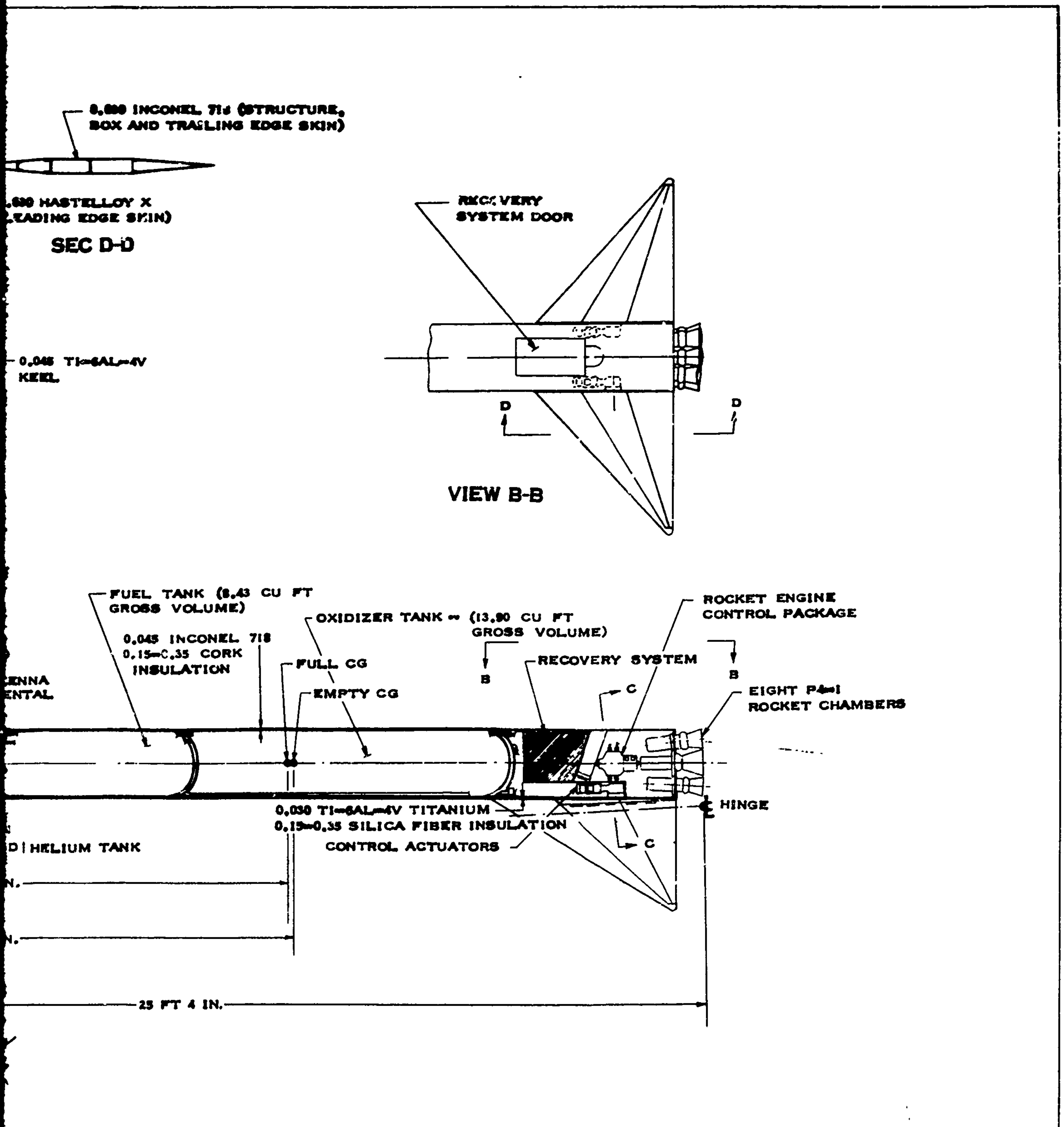


FIGURE 7. PARAWING VEHICLE, INBOARD PROFILE

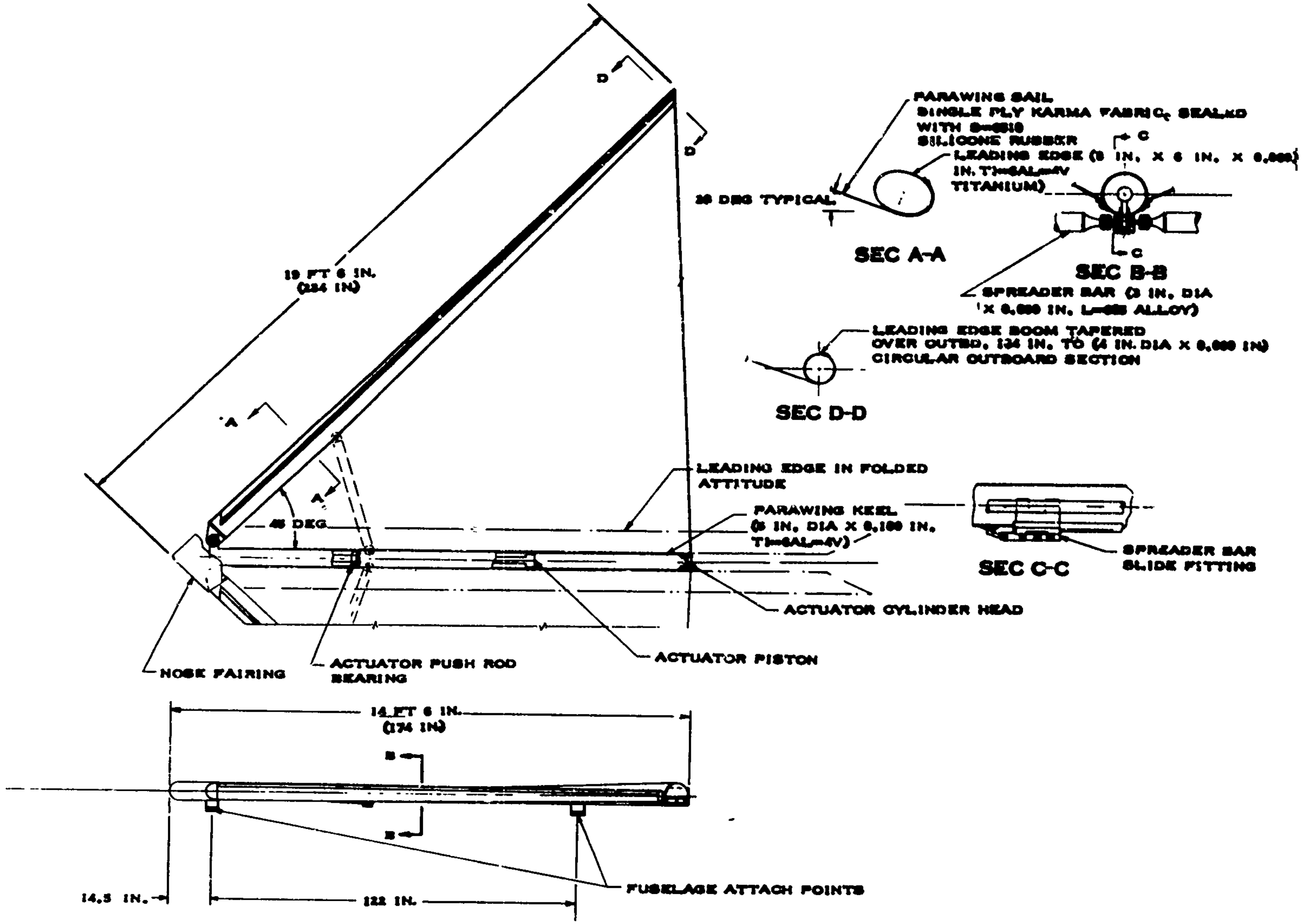


FIGURE 8. PARAWING ASSEMBLY

3.3 PROPULSION SYSTEM

3.3.1 System Description

The proposed propulsion system for the HI-HICAT vehicle utilizes, wherever possible, components of the AR-2 and the LR64-NA-4 (Rocketdyne Model P4-1) propulsion system or modifications thereof. The LR64-NA-4 rocket engine is used in the Navy AQM-37A target missile and has completed over 300 flights. The propulsion system consists of a cluster of eight LR64-NA-4 regeneratively-cooled thrust chambers with increased area expansion ratio and a storable propellant feed system. Modifications to some of the control components is necessary to maintain low pressure drop in the propellant feed system with the required higher propellant flow rates. Figure 9 is a schematic of the complete propulsion system for the lifting body vehicle. The weight penalty associated with highly-pressurized, noncylindrical propellant tanks is too severe for the lifting body vehicle and a turbopump propellant feed system has been added. The propellants are IRFNA and MAF-4 (oxidizer and fuel, respectively), and the pressurant is gaseous helium. The propulsion system is divided into two modules, the rocket engine assembly module and the propellant feed system module.

The integrated rocket engine assembly module consists of a cluster of eight thrust chambers, an engine mount, two boost valves (one each, oxidizer and fuel), two sustainer valves, a propellant manifold, an engine electronic control package, and an electrical cable assembly. The module is identical for either the parawing or the lifting body system except that a throttle valve is added to modulate the sustainer engine on the parawing vehicle. The throttle for the lifting body vehicle is part of the turbopump assembly and modulates the sustainer engine indirectly by modulating the turbopump.

The integrated propellant feed system module for either vehicle includes a fuel and oxidizer tank assembly, two propellant valves, two burst diaphragms, a pressure relief valve, and a start valve. For the lifting body vehicle, the propellant feed system module also includes a turbopump assembly which consists of a turbopump, a gas generator, two gas generator valves, and a gas generator throttling valve.

Propulsion system operation for the lifting body will be conducted in the following sequence:

1. System Arming. Upon receipt of the arming signal from the aircraft, the electronic package will arm the firing circuits. After the firing circuits are armed, a signal will be sent to the aircraft to indicate that the system is armed and the propulsion system is ready for vehicle launch.
2. Propulsion System Pressurizing. After launch the start valve will be actuated, gaseous helium will flow through the pressure regulator and check valves, rupture the burst diaphragms, and pressurize the propellant tank.
3. Propulsion System Start. Upon receipt of the propulsion start command signal, the main propellant valves are actuated and propellants flow to the gas generator. One second after the turbopump starts to accelerate, the booster and sustainer propellant valves are actuated and propellants flow to all thrust chambers. The time required to achieve 90 percent rated thrust is two to four seconds after actuation

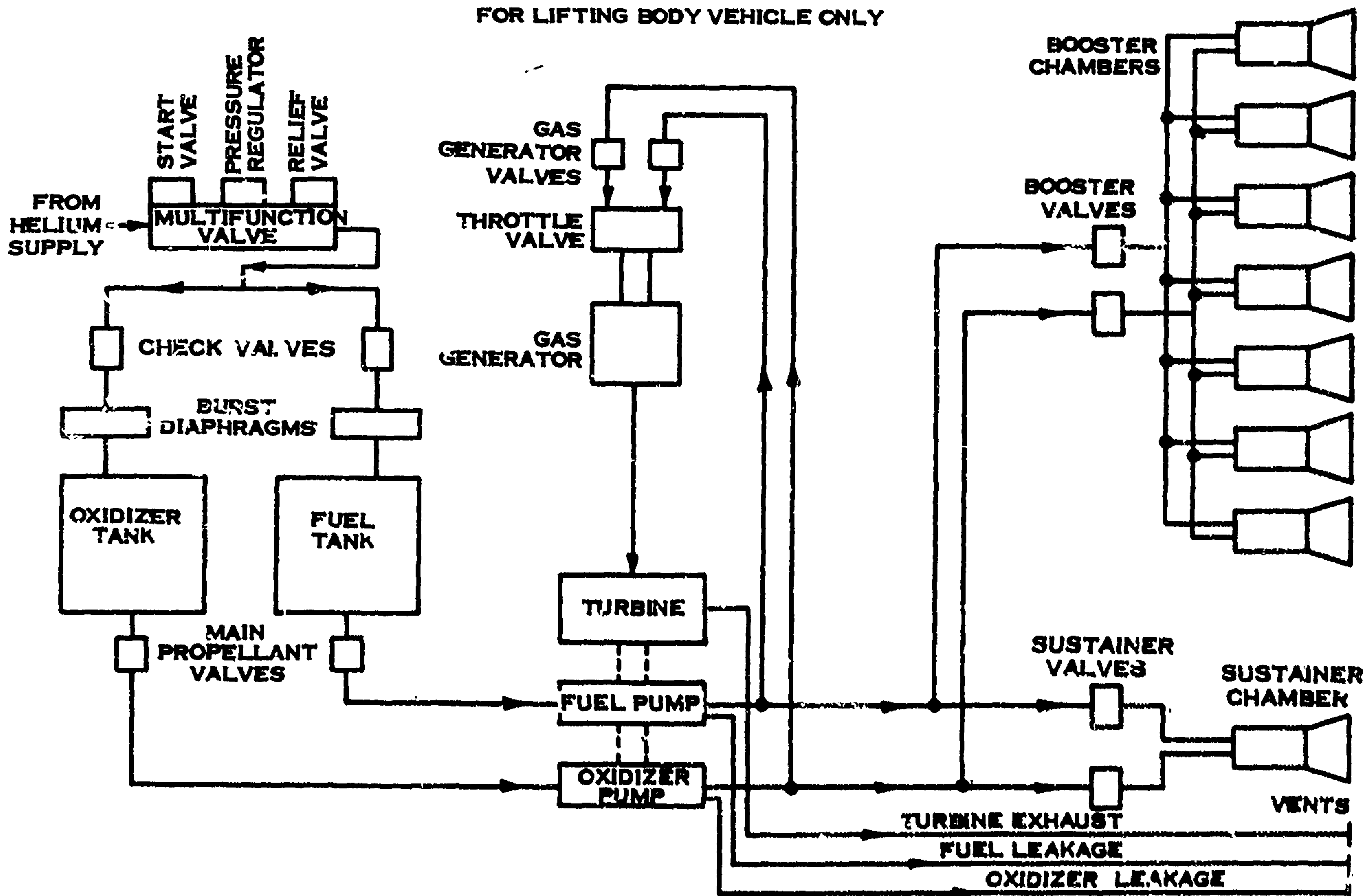


FIGURE 9. PROPELLANT AND PROPULSION SYSTEM

of the start valves. The gas generator throttling valves and the control system will regulate power to the turbine. For the pressure-fed parawing propulsion system, the start sequence consists only of actuating the propellant valves.

4. Boost Termination. When the desired boost operating time has been achieved, a boost termination signal will be received from the vehicle flight control system by the electronic control package. The electronic control package will close the boost valves. The thrust from the booster thrust chambers will decay to zero and the sustainer thrust chamber will continue to operate at the thrust level commanded.
5. Sustainer Termination. After boost termination, the sustainer thrust chamber will continue to operate throughout its thrust range as commanded by the vehicle flight control system. The sustainer will operate until propellant depletion or until commanded to terminate by the vehicle flight control system. If commanded to terminate, the sustainer and gas generator valves will close.
6. Purging Sequence. After all the thrust chambers have shut down and prior to the initiation of the parachute recovery sequence, the electronic control package initiates and controls a purging sequence. First the pressurizing start valve is closed and the fuel oxidizer valves are opened for a time sufficient to empty the tank and plumbing and to reduce the pressure in the tank to one atmosphere. After the oxidizer valves close, the fuel system is purged in a similar manner.

3.3.2 System Performance

Nominal performance summaries, Figures 10 to 13, have been prepared for the propulsion system operating during the boost phase and during the sustain phase. It will be shown in Section 5.2.1 that eight rocket chambers result in near optimum boost performance. Only one sustainer is considered in the present analysis but two may be required for cruising at very high or very low HI-HICAT altitudes.

Rocket engine performance was determined from P4-1 production verification tests. These tests, which simulate AQM-37A target missile flight mission duty cycle at ambient conditions, have been performed on 62 P4-1 propulsion systems. Booster performance generated during those tests was extrapolated to altitude conditions with the larger expansion ratios proposed for this HI-HICAT system. (The P4-1 booster has a 6.4:1 expansion ratio.)

Specific impulse variations during sustainer rocket engine throttling are small. In the parawing system the throttle valve will be designed to maintain a constant mixture ratio to the thrust chamber throughout the throttling range. Theoretically, specific impulse will increase only 2 seconds for mixture ratios from 2.8 to 3.2, and will increase only 1 second for chamber pressures from 75 to 300 psia. In the lifting body system, the same throttling valve will be used to provide a mixture ratio of 0.08 to the turbopump gas generator and to control sustain thrust as commanded by controlling propellant flow to the gas generator. Thrust chamber mixture ratio then is controlled by fixed calibration devices downstream of the propellant pump.

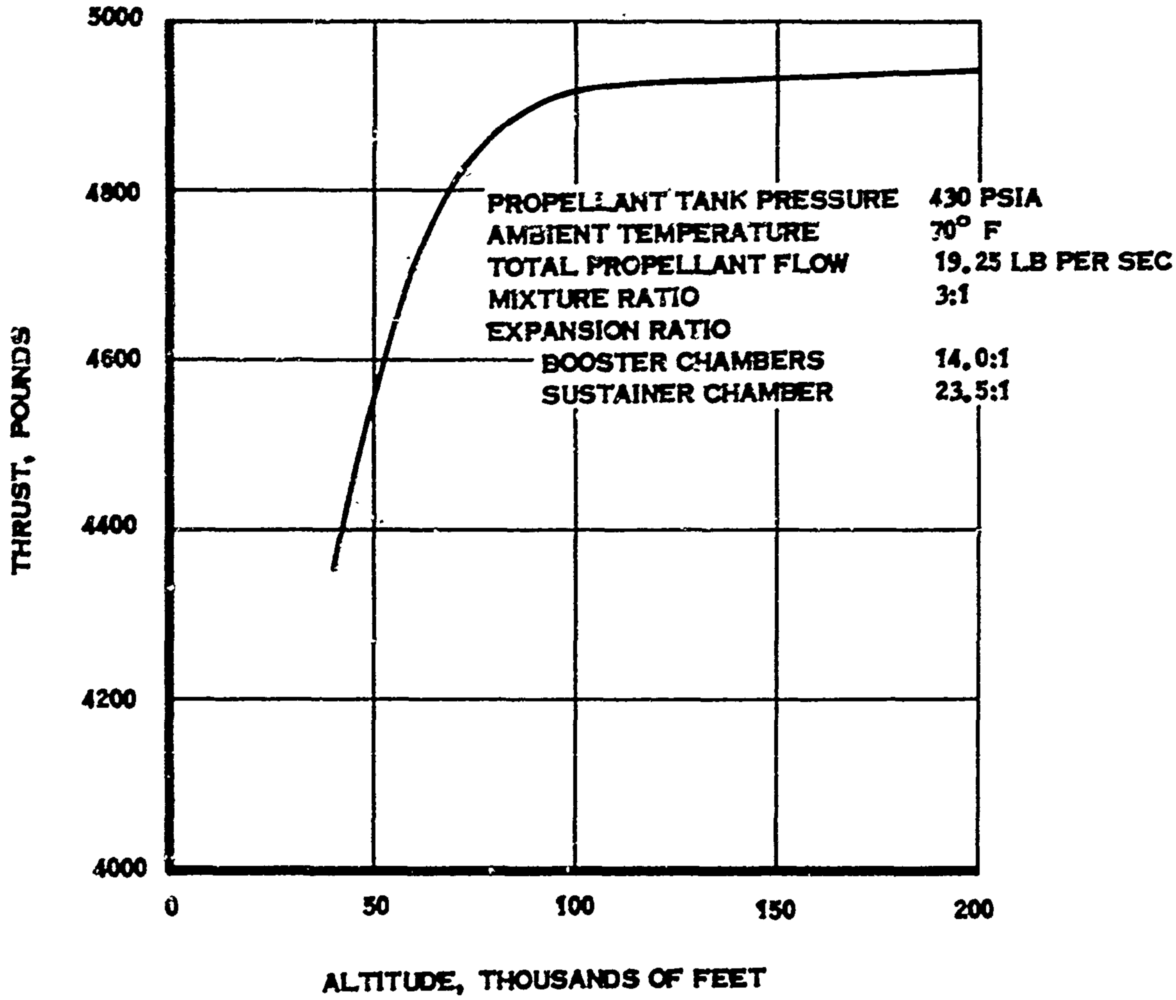


FIGURE 10. PARAWING VEHICLE BOOSTER PERFORMANCE

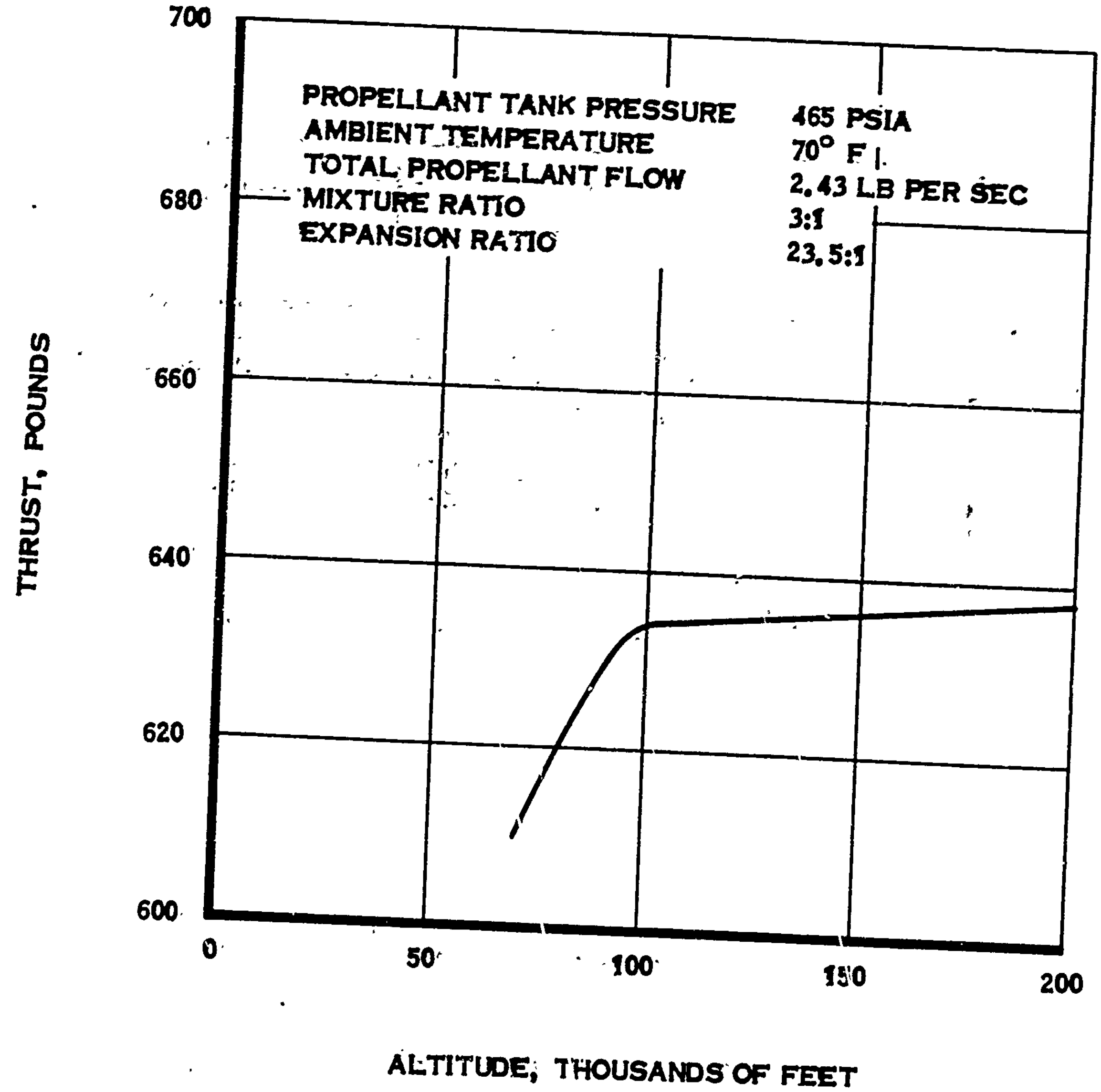


FIGURE 11. PARAWING VEHICLE SUSTAINER PERFORMANCE

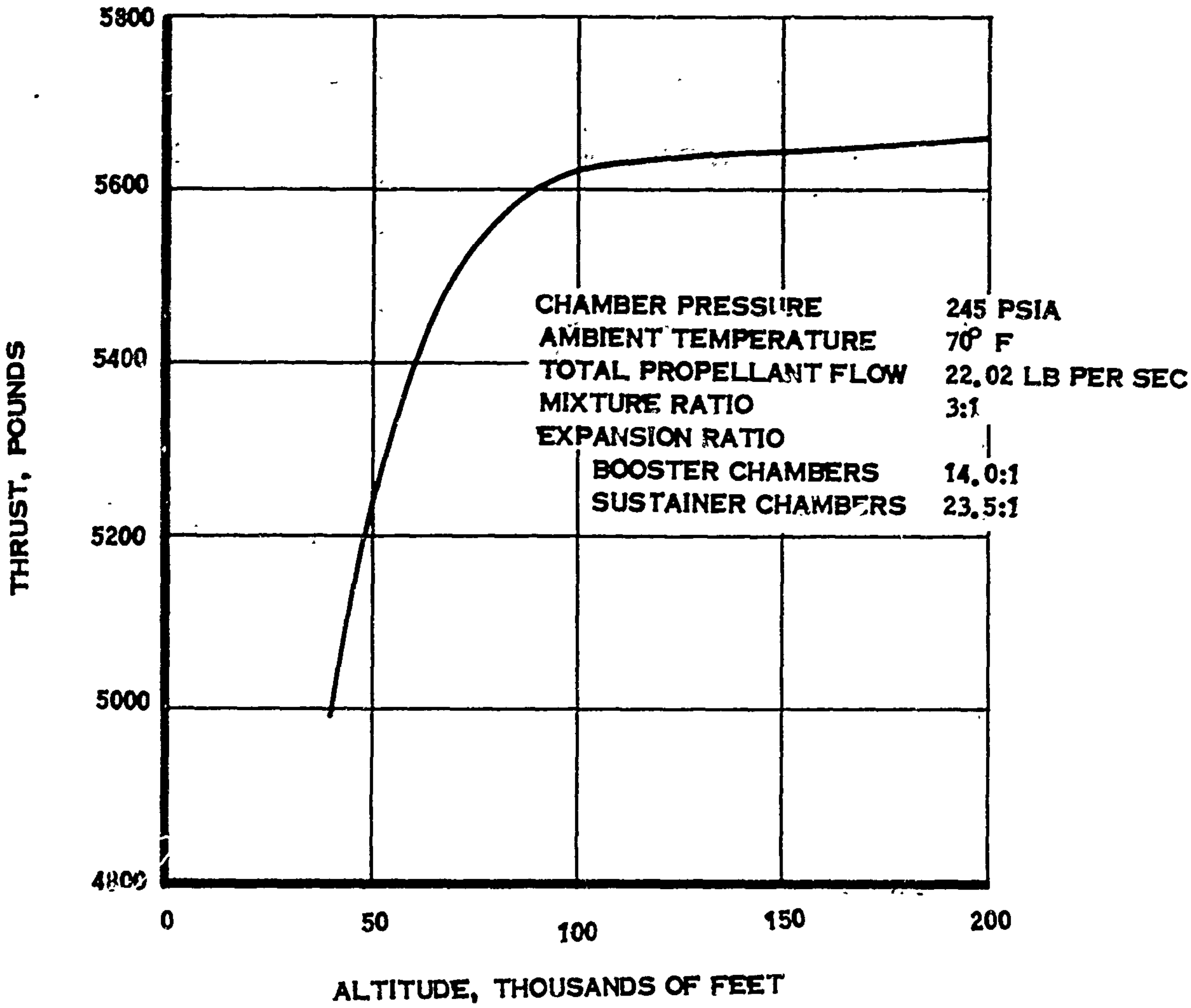


FIGURE 12. LIFTING BODY VEHICLE BOOSTER PERFORMANCE

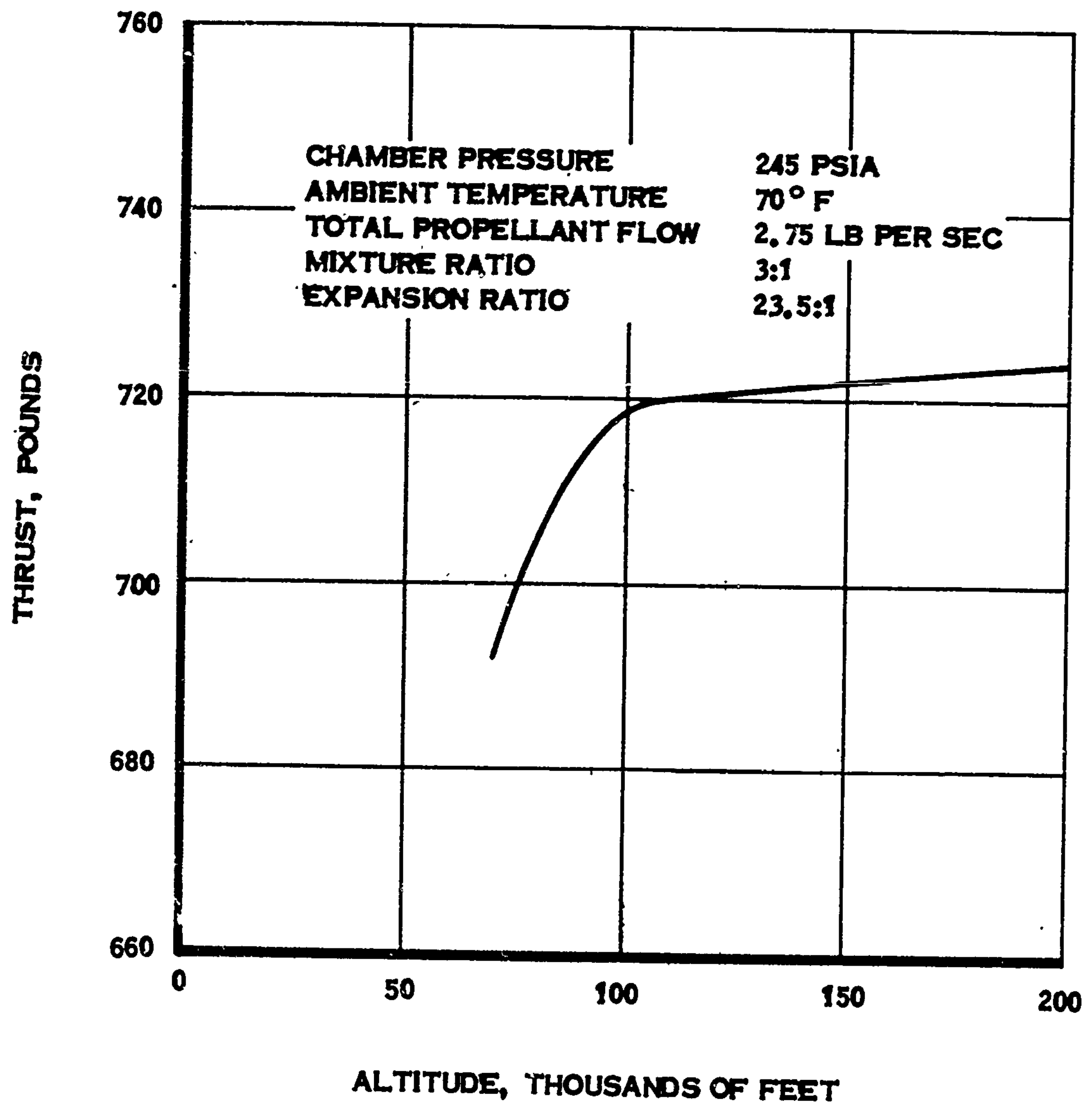


FIGURE 13. LIFTING BODY VEHICLE SUSTAINER PERFORMANCE

The propellant flow required for the propulsion system with a turbopump installed is given below:

Thrust	Number of Chambers Firing	Main Propellant Flow Rate (3.0 Mixture Ratio)	Gas Generator Flow Rate (0.08 Mixture Ratio)
5660 lb	8	22.02 lb per second	0.622 lb per second
595	1	2.280	0.2905
340	1	1.303	0.0776

During sustainer engine throttling, the rocket chamber pressure can reduce to 86 psia at 250 pounds of thrust. The chamber injector inlet pressure at this thrust level is 115 psia. Referring to Figure 14, the maximum propellant temperature at this pressure is 250°F and 335°F for the oxidizer and fuel, respectively. If the propellant temperatures exceed these temperature limits, propellant boiling will occur and the propellant injection into the thrust chamber will be two phase (gas and liquid). The two-phase propellant injection will result in combustion instability or chugging. The oxidizer will experience a 110°F temperature rise through the cooling jacket. Therefore, the maximum propellant inlet temperature of the rocket engine assembly must be maintained at less than 140°F. The fuel temperature is not as critical to combustion instability as the oxidizer. However, if the fuel temperature is allowed to vary significantly so that there is a large differential temperature, the mixture ratio will vary and will result in the premature depletion of one of the propellants.

3.4 LAUNCH SYSTEM

The launch sequence proceeds by the pilot entering the launch maneuver at a given speed, altitude, heading, and geographic location. He then pulls up the aircraft through the preset launch attitude angle at which point the HI-HICAT vehicle is released in a stable attitude from the launch pylon with the guidance and control equipment in operation. The launch pylon will contain jettisoning equipment only if detail analysis or tests show inadequate separation. At this point, a trapeze arrangement which drops down upon receipt of the launch command appears to be adequate. Propellant tank pressurization and propulsion system operation begin only when the vehicle is separated from the parent aircraft by a distance of at least 100 feet.

The F-4C is the preferred launch aircraft. It has a large store carrying capacity and considerable excess thrust in the supersonic flight regime as compared to other supersonic fighters in the Air Force and Navy inventory with the possible exception of the YF-12A long range interceptor, an aircraft whose performance is subject to extreme security and whose availability is uncertain. A standard centerline store for the F-4C is a 600 gallon tank which is easily accelerated to high supersonic speeds. The drag and weight of the recommended HI-HICAT vehicle is less than for this store, which assures the adequacy of the F-4C store carrying abilities. Figures 15 and 16 are drawings showing ground and aircraft clearances with the proposed HI-HICAT vehicles installed. Note that the vertical fin of the parawing vehicle must be folded to provide ground clearance. The hinge line is canted so that aerodynamic forces can force the fin into a down and locked position before launch.

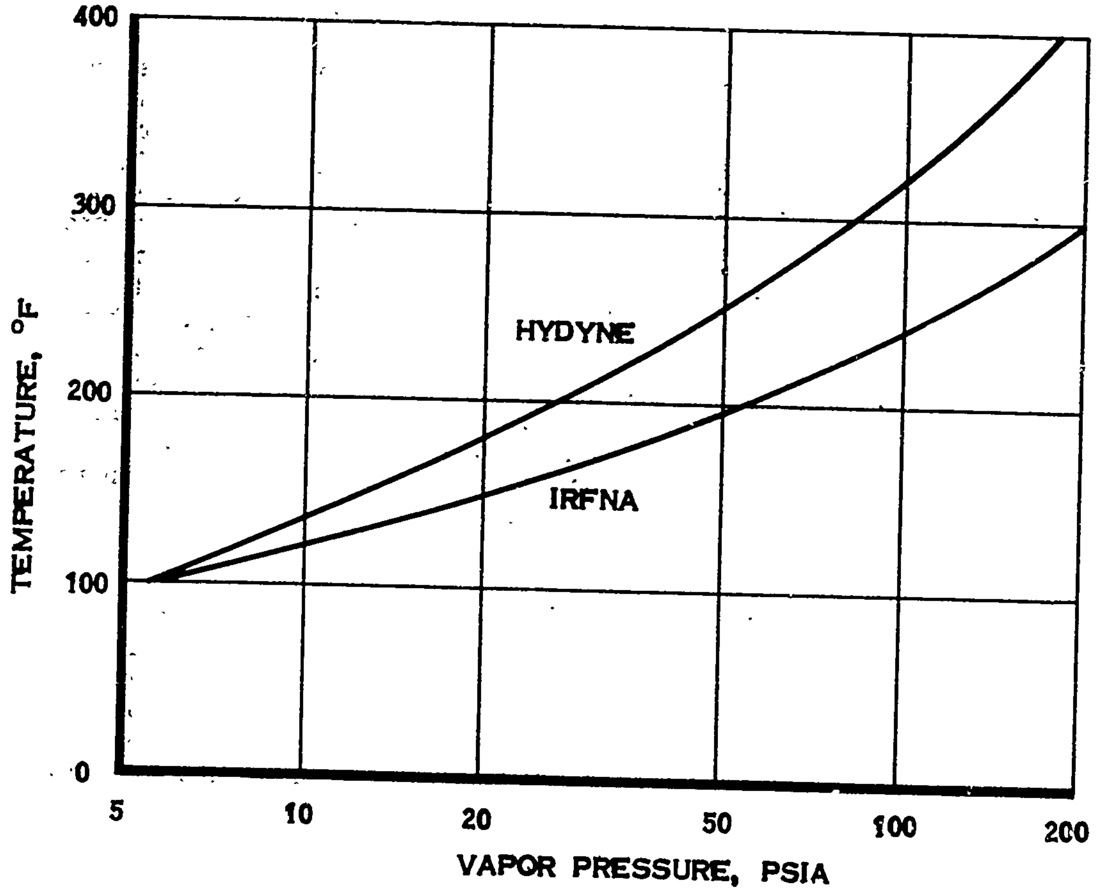
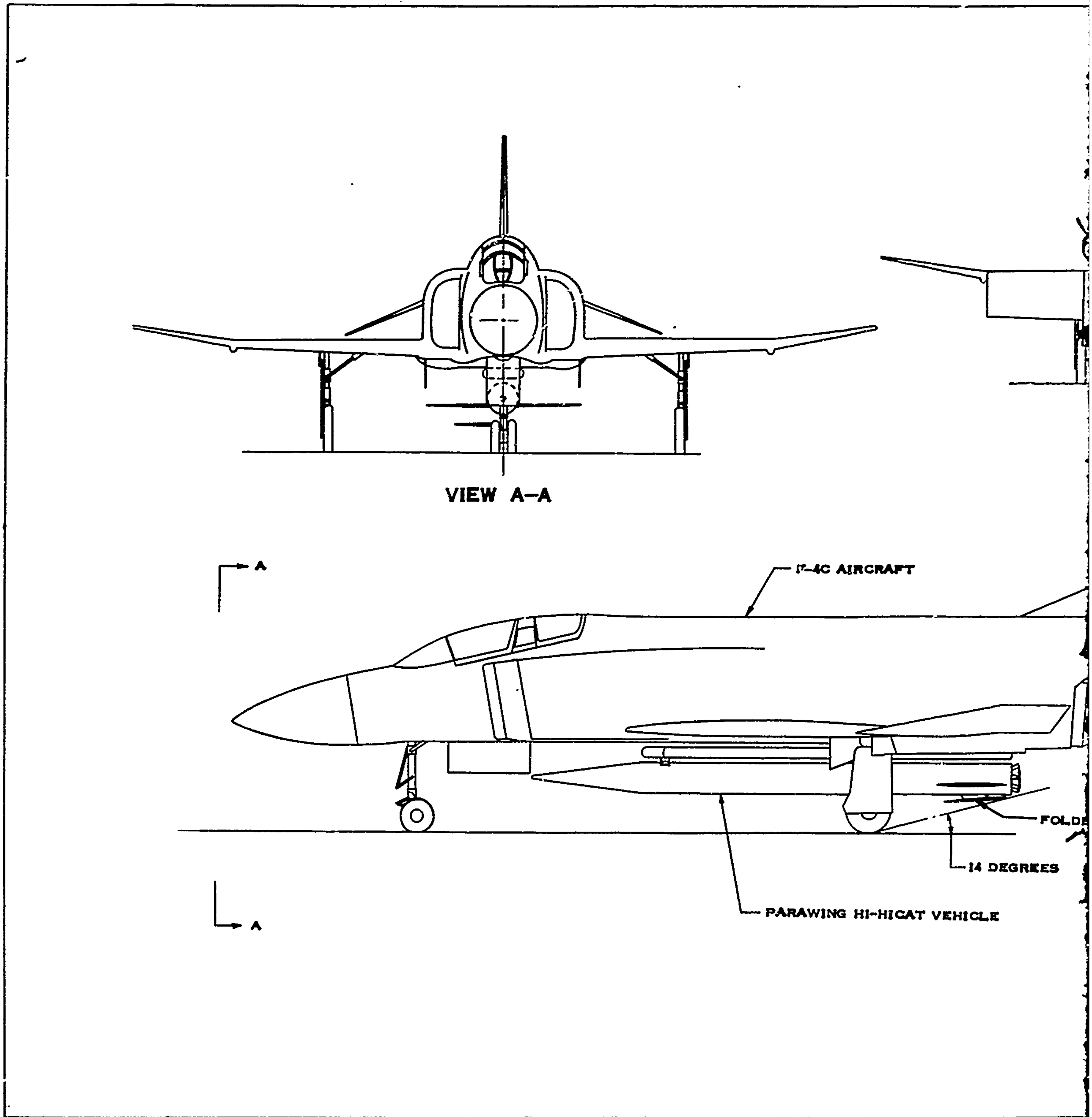


FIGURE 14. VAPOR PRESSURE


VIEW A–A
A
F-4C AIRCRAFT
FOLDE
14 DEGREES
PARAWING HI-HICAT VEHICLE
A

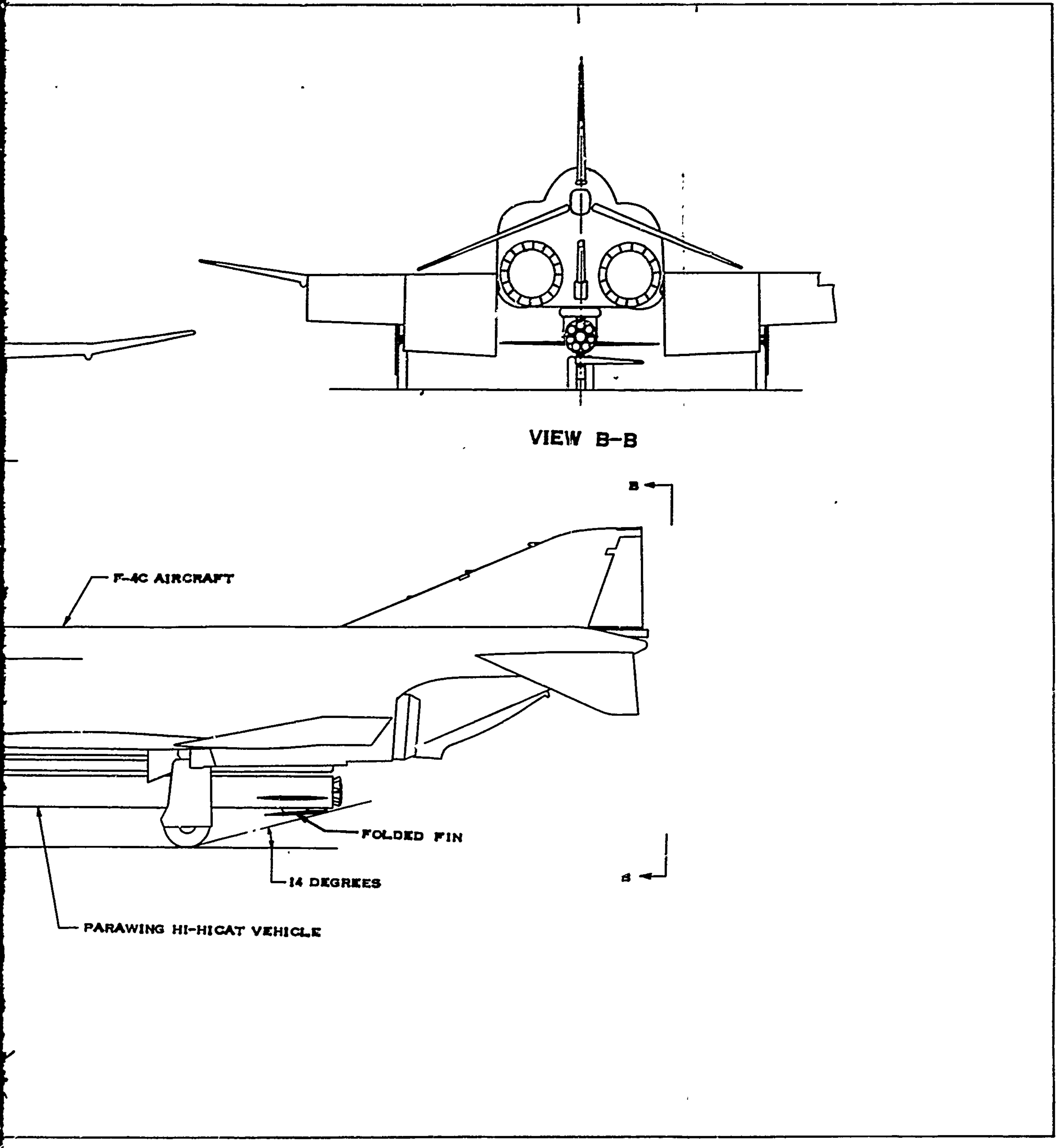
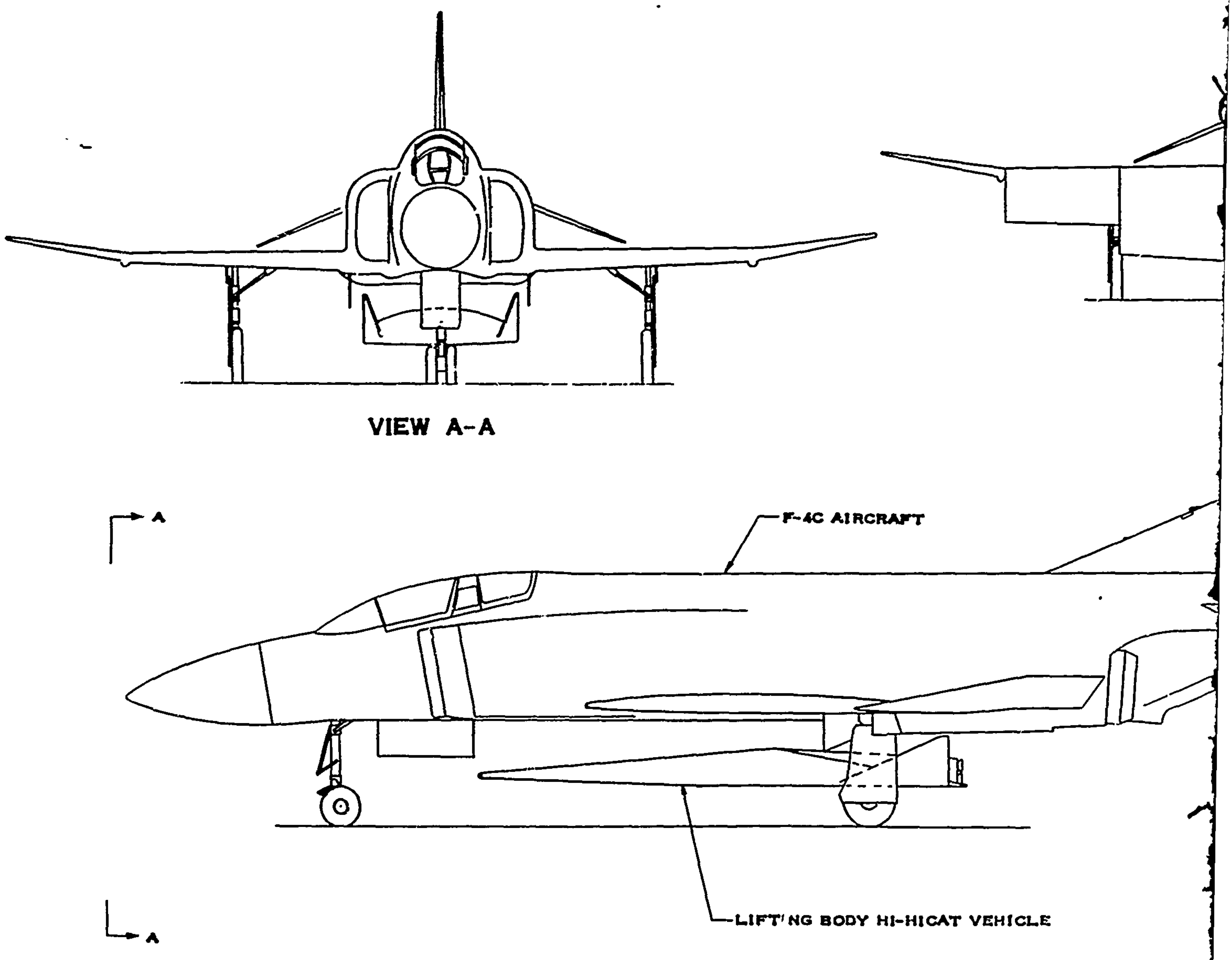


FIGURE 15. PARAWING VEHICLE INSTALLATION ON F-4C



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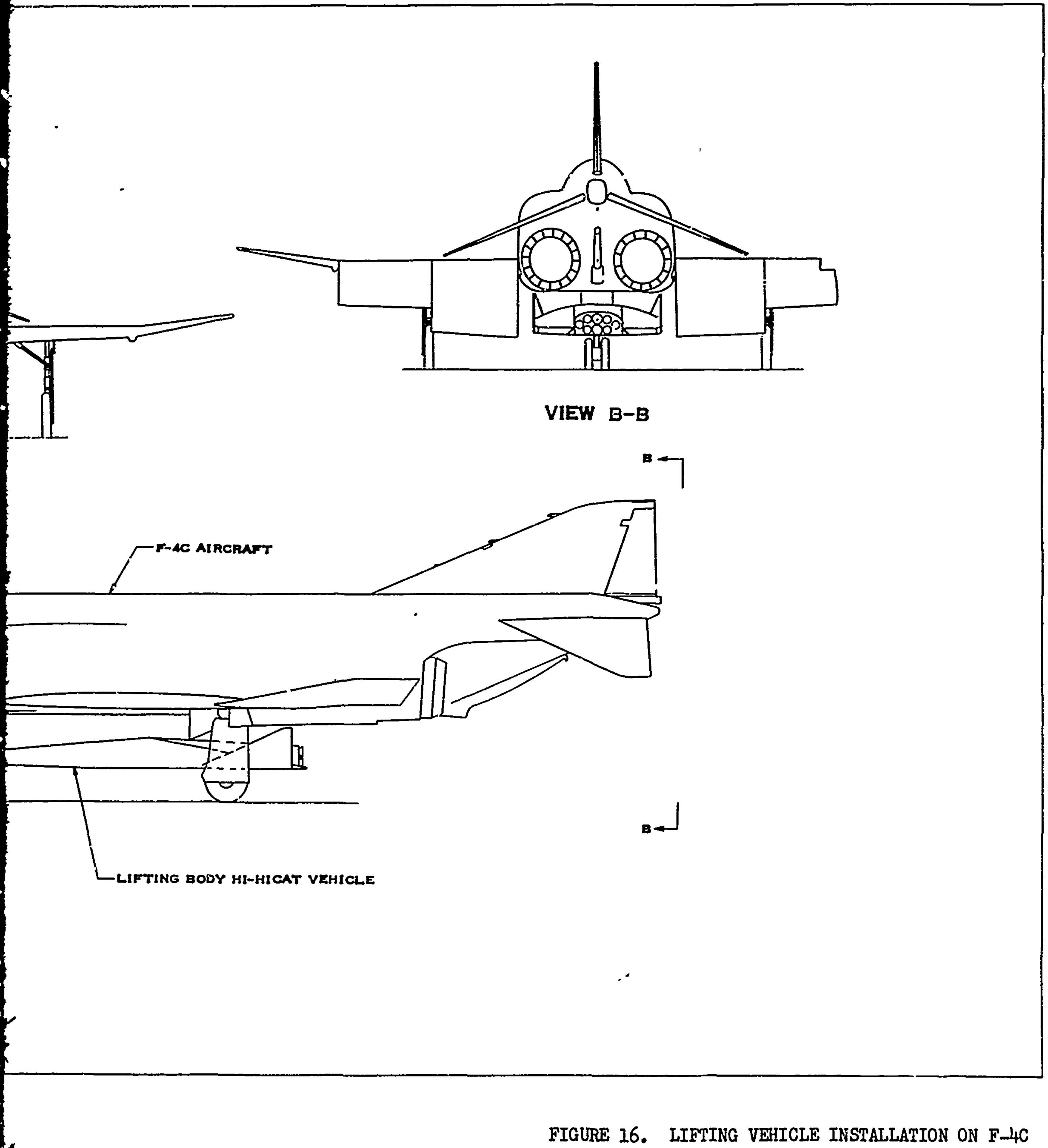


FIGURE 16. LIFTING VEHICLE INSTALLATION ON F-4C

A larger HI-HICAT vehicle with a somewhat greater range could have been designed for a supersonic launch from the B-58 bomber. The cost of operating this aircraft eliminated it from further consideration. Studies completed by Booz-Allen Research, as reported in Reference 4, generated cost figures for a B-58 bomber operating as a launch platform for a high altitude probe. The probe has a Castor rocket as a first stage and was launched at a speed of 2000 fps and an altitude of 40,000 feet. Spread over 60 launches the cost for aircraft maintenance, fuel, oil, crew, and aircraft modifications was $14,000 per launch, an excessive figure.

The F-4C is normally equipped with the devices which are desirable or necessary for an air launch. The aircraft has an inertial navigation and attitude reference system which consists of two separate gyro reference components and a navigation computer. The inertial navigation set AN/ASN-48 is the primary azimuth and attitude reference device; it also supplies direction, velocity, and distance inputs to the navigation computer AN/ASN-46. The AN/AJB-7 is the standby attitude reference component and also supplies information for various bombing maneuvers. The navigation computer receives inputs from either the inertial navigation set or the air data computer. The navigation computer system makes the following computations during flight:

1. Present aircraft latitude and longitude.
2. Aircraft ground track angle relative to magnetic heading.
3. Aircraft great circle distance from base.
4. Aircraft great circle bearing to target or alternate base.
5. Aircraft ground speed.

The bombing computer function of the AN/AJB-7 includes a release angle control which is applicable to launching the HI-HICAT vehicle. The low angle control can be set to angles from 0 to 90 degrees and will automatically release the store at the preset angle.

3.5 RECOVERY SYSTEM

3.5.1 Design Factors

The HI-HICAT vehicle is designed to be recovered. Drone type vehicles are sometimes expendable when simple on-board equipment and high production rates permit low unit costs. Unfortunately, the conditions favoring expendability do not apply to the HI-HICAT system. Another variation of the expendability concept is to recover only the nose cone unit with the instrumentation and guidance equipment installed. The recovery package would weigh less, but otherwise the recovery requirements would remain the same. This concept of partial expendability is also rejected since the cost saving in reusing all systems is believed to outweigh the disadvantage of increased recovery weight.

Another design decision which was faced was to choose between a parachute recovery or a glide-down followed by a horizontal landing. Listed below are the disadvantages of each technique compared with the other.

Horizontal landings:

1. A flare is required just before touchdown to reduce the rate of descent

to a reasonable value. Since no thrust is available, the approach and final would have to be carefully and precisely executed. These difficult maneuvers would require a guidance and control system of greatly increased complexity.

2. The long glide-down at subsonic speeds requires additional weight in the power supply for the tail actuators.
3. Configuration compromises are required to achieve proper trim capabilities and stability for a vehicle flying at both subsonic and supersonic speeds.
4. Overwater or remote operations would introduce entirely new landing problems and recovery risks.
5. Added weight and complexity of landing gear.

Parachute recovery:

1. Net overall weight is higher.
2. Air snatch expensive and risky to personnel and aircraft.
3. Parawing must be jettisoned and may be lost (if installed).

The disadvantages for a horizontal landing are considered to outweigh those for a parachute recovery.

Aerial recovery has been developed recently to a fair degree of reliability. Surface recovery from land or sea would require additional equipment for landing shocks or flotation and search aids. This additional weight would be small and a saving in recovery costs may be possible. The development schedule includes tests for both types of recovery and an analysis after these tests as to the preferred technique. For the purpose of this preliminary design study, the parachute has been designed for aerial recovery with one of the standard systems available such as on the C-130 aircraft and a chaff package has been added as a localization aid for the recovery aircraft.

The parawing design calls for the wing to be jettisoned before the drogue chute is ejected. The HI-HIGH vehicle cannot be loaded aboard the recovery aircraft with the parawing installed. The wing actuator would be locked open and the wing would float or glide down to the ground to be recovered, refurbished and reused.

The coordinates of the HI-HIGH trajectory must be established accurately in order to assure that the recovery aircraft is stationed at the proper latitude and longitude. The launch aircraft must enter the launch maneuver at a known latitude, longitude, and flight direction; a condition which will be readily met after practice and with the aid of the navigation computer in the launch aircraft. Performance calculations will give the expected range. If the detailed design studies indicate an uncertainty in the length of time that the cruise engine will operate before propellant exhaustion, then a guidance function shuts down the engine at a preselected range slightly less than maximum.

3.5.2 System Description

The recovery sequence will be initiated when the vehicle has decelerated to Mach 1.5. This event will occur at an altitude of 70,000 to 90,000 feet. Under these conditions the recovery system configuration will be governed primarily by the aerial pickup requirement. Space limitations call for

components of high specific performance, thereby limiting the use of hardware designed for other purposes. An aerial pickup parachute of advanced design is proposed because test experience with this design has shown better performance and greater reliability than with other such systems currently in use or under development. This design is still undergoing development under an Air Force contract for a "Universal Aerial Recovery System", but it has exhibited satisfactory characteristics to date.

A mortar-deployed drogue chute of proven design is proposed for deployment of the main parachute system at an altitude of 45,000 feet. The initial conditions of Mach 1.5 at altitudes of 70,000 to 90,000 feet enable a relatively light-weight and efficient ribbon drogue design to be utilized. The maximum dynamic pressure at deployment would be less than 150 psf. The recovery sequence is depicted in Figure 17 (for the parawing vehicle).

Parachute deployment and opening loads are to be taken at hard points located far enough aft of the vehicle center of gravity to minimize bending loads and vehicle gyrations. After the system has stabilized in steady descent, the transfer of the main parachute harness attachment to a hard point well forward of the center of gravity is necessary to assure good vehicle towing stability during the aerial recovery operation. Further analysis is needed to determine if these hard points can be conveniently located on one side of the vehicle, or whether a less desirable nose-cone position will have to be used to hold the vehicle in a stable aerodynamic trim attitude suitable for both stable deceleration and acceleration. Because the unsymmetrical empennage favors a canted trim attitude, the feasibility of the side attachments appears good and their use is assumed for purposes of this study.

The drogue pack is ejected through an opening in the main compartment cover. The deployment bag is stripped off the canopy by the momentum obtained from the mortar firing augmented by the aerodynamic drag. The drogue chute drag is sufficient to decelerate the vehicle to a terminal dynamic pressure of approximately 90 psf at 45,000 feet for main parachute deployment.

Deployment of the main chute system is effected by release of the drogue chute. The main chute is annular with a 65 percent vent and a ringslot pickup chute is positioned above the main chute vent. The main compartment cover is retained by the drogue release mechanism and functions as a link in the bridle between the drogue riser and the main deployment bag. The main pack is extracted from the compartment by drogue chute drag. The reliability of this operation can possibly be enhanced by also employing an ejector bag under the main pack. However, use of an ejector bag is not proposed on the assumption that a detailed design study would show the final pack-compartment geometry provides a satisfactory exit path without it.

The reefing system consists of a skirt reefing line with four second line cutters on the pickup canopy and a main canopy skirt reefing line with eight second line cutters. Thus the pickup canopy functions as a two-stage drogue to bring the system down to a low dynamic pressure (about 5 psf) at the time the main canopy is disreefed to open fully. This permits a very lightweight main canopy structure to be employed (with suspension line strength determined by the maximum pickup load).

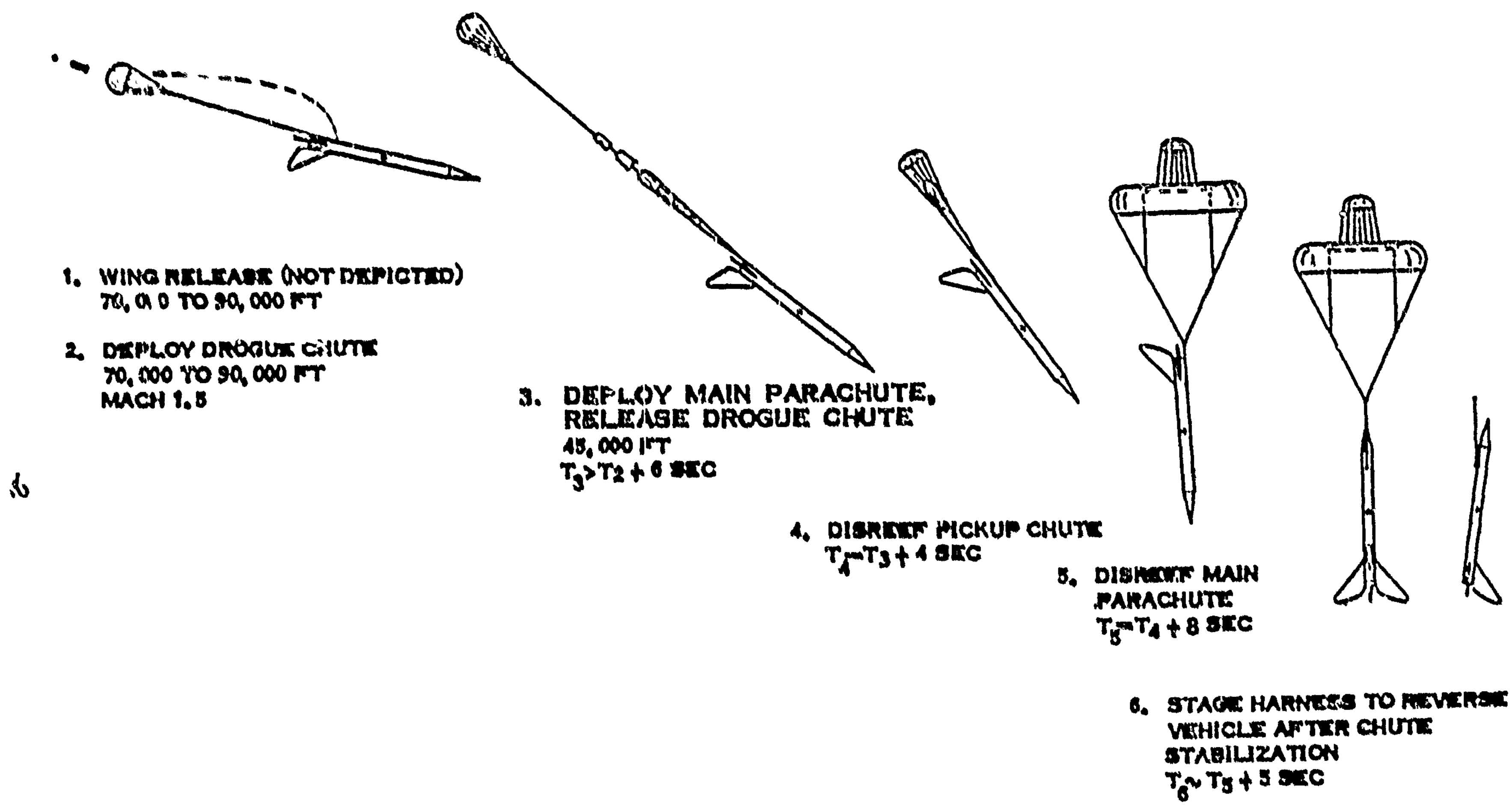


FIGURE 17. RECOVERY SEQUENCE

The sequence controller will perform the following functions for the recovery system in the manner stated:

1. Arm the deployment system after the vehicle has passed to an altitude above the recovery initiation altitude by receipt of a signal from the HI-HICAT guidance and control system.
2. Provide sensing of the predetermined deployment initiation altitude with a baroswitch.
3. Provide an electrical signal to the wing release device.
4. Provide an electrical signal to the drogue mortar pyrotechnic, initiating it shortly after the wing release is effected.
5. Provide a signal for activating the chaff package.
6. Provide a six-second time delay after the drogue deploy signal to arm the 45,000-foot baroswitch. This delay assures a safe level of dynamic pressure for the main deployment when the baroswitch is armed.
7. Provide sensing of the 45,000 foot altitude point with a baroswitch.
8. Provide an electrical signal to the drogue disconnect device pyrotechnic, initiating it, releasing the main parachute compartment door and deploying the main parachute assembly.
9. After parachute deployment, the pickup chute reefing line cutter will actuate, disreefing the pickup chute.
10. Cut the main chute reefing line after the pickup chute disreef, allowing full inflation of the main chute assembly.
11. Provide an electrical signal to the main chute aft section release device, actuating it and transferring the parachute attachment to the forward attach point.
12. Provide an electrical signal for disconnecting the parachute, should the landing occur without a successful aerial pickup.

The sequence controller utilizes extremely rugged and reliable pyrotechnic switches to achieve both the electrical switching and sequential time delays. To further enhance reliability, a dual system is employed such that no single malfunction will result in a system failure.

The power supplies consist of sealed rechargeable nickel-cadmium batteries, one for each of the redundant sequence controllers to assure dependable operation in a space environment, exact recharging capacities, and long shelf life.

SECTION 4

STRUCTURAL ANALYSIS

4.1 THERMODYNAMICS

4.1.1 Parawing Vehicle

A rocket powered vehicle must be flown at high speeds to maintain flight at extreme altitudes. High speeds, however, result in a multitude of thermodynamic problems. Table 2 indicates the rapid rise in skin temperature as the cruising Mach number is increased. These temperatures are based on radiation equilibrium for an insulated surface with an emissivity of 0.8, and they would be closely approached during the cruise. It is evident that there will be a tradeoff between a high speed for longer ranges and a low speed in order to reduce the thermodynamics problems to a minimum.

Hot spots will exist at points between the fuselage and the parawing where the radiation viewing angles are restricted. Unfortunately, the temperature environment for complex shapes is difficult to analyze. "Hot" tunnel tests will be required for a parawing vehicle which may not be necessary for a lifting body vehicle.

The design philosophy being followed is that heat shield material will be used on a few critical areas, especially the titanium leading edge booms for the parawing and the hot spots between the fuselage and the parawing. This material can be either passive insulation or an active material which undergoes chemical or physical changes such as sublimation. Detail calculations relating the thickness of the heat shield materials to the cruise speed have not been attempted in this study and are best delayed until the development of a HT-HICAT vehicle.

To aid the analysis of the pressurization and cooling system, the temperatures along the fuselage nose cone were detailed for the most adverse cruise condition that is likely to be encountered. The parawing, when extended at the beginning of cruise, may orient the vehicle at an angle of attack as high as 40 degrees at 200,000 feet at Mach 6. The nose cone skin temperatures for this condition are given in Figure 18 where the heating rate was assumed to vary radially according to Lee's laminar heating distribution.

4.1.2 Lifting Body Vehicle

The radiation equilibrium temperatures for a lifting body are presented in Figure 19 for Mach 3.0 at 70,000 feet and Mach 8.4 at 175,000 feet which represent two extremes in the flight conditions. It is evident that the temperatures are extremely high at Mach 8.4 and that this speed is not a practical cruise condition. However, compared to a parawing vehicle, the lifting body vehicle has advantages which would permit higher cruise speeds. For one, it is a configuration for which a large amount of test data is available. Hence a greater degree of certainty can be established for the magnitude of the thermal environment. For another, there are no hot spots such as exist between the parawing and the fuselage. And finally, the body shape is more compact and it can be protected more readily with heat shield materials.

TABLE 2

HI-HICAT PARAWING VEHICLE

RADIATION EQUILIBRIUM TEMPERATURES

Altitude	70,000 Ft			130,000 Ft			200,000 Ft		
Mach	3	4	5	4	5	6	5	6	7
Angle of Attack	5°	5°	5°	25°	25°	24°	30°	30°	30°
Recovery Temperature	640°F	1160°F	1770°F	1370°F	2050°F	2810°F	2100°F	2850°F	3510°F
Q-Ball	620	1090	1610	1100	1530	1950	1170	1470	1760
Parawing Leading Edge	590	1010	1460	950	1300	1630	950	1180	1420
Stabilizer Leading Edge	630	1120	1670	1210	1710	2220	1400	1760	2120
Vertical Fin Leading Edge	620	1100	1610	1170	1650	2130	1340	1680	2020
Fuselage Bottom									
X = 5 Ft	525	860	1200	540	710	880	490	610	720
X = 10 Ft	515	850	1180	490	640	800	450	540	650

* X=Longitudinal distance from nose

GEOMETRIC DATA

	Diameter	Sweep
Q-Ball	2.5 In.	—
Parawing Leading Edge	4 (minimum)	45 Deg
Stabilizer Leading Edge	0.15	45
Vertical Fin Leading Edge	0.3	60
Body	19.0	—

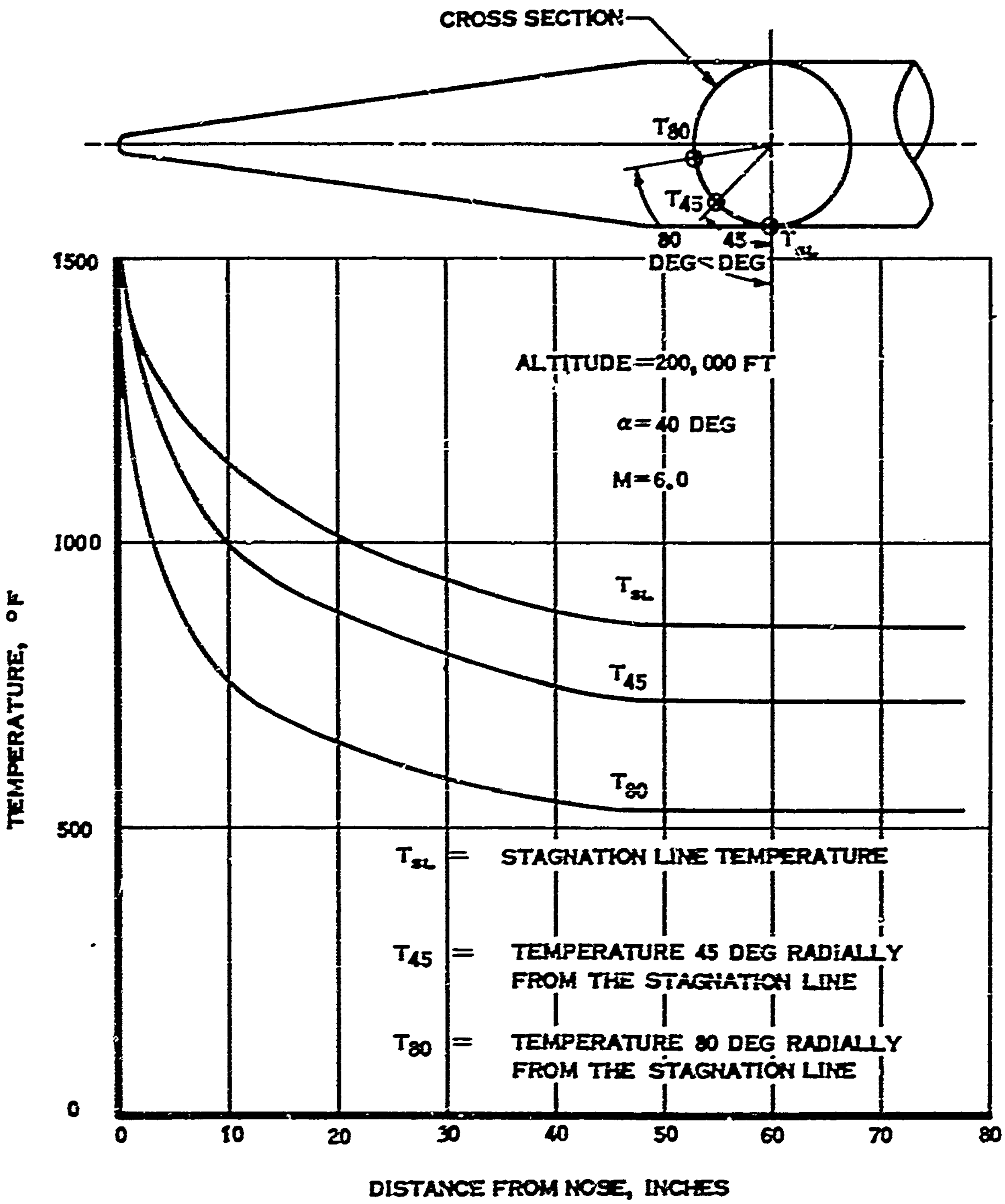


FIGURE 18. NOSE CONE RADIATION EQUILIBRIUM TEMPERATURES

CONDITION	A	B
MACH	3.0	8.4
ALTITUDE	70,000 FT	175,000 FT
ANGLE OF ATTACK	7 DEG	30 DEG
EMISSIVITY	0.8	0.8

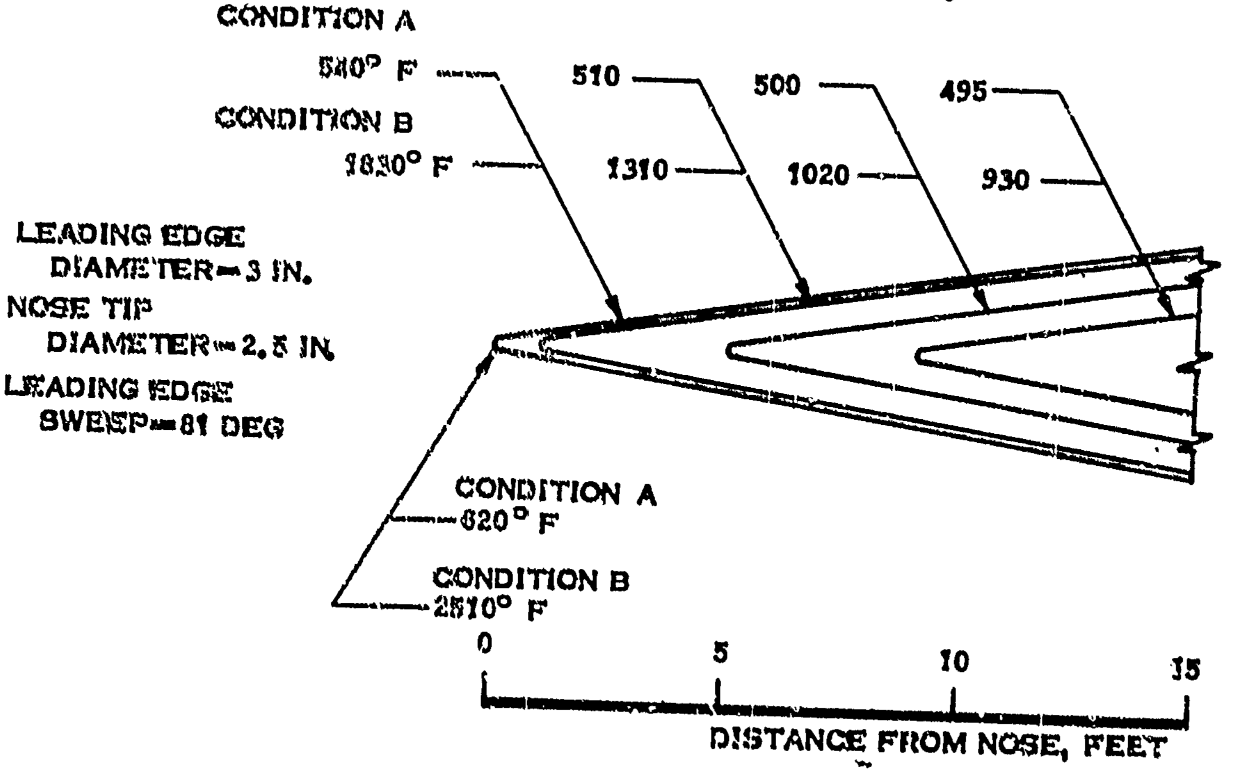


FIGURE 19. LIFTING BODY VEHICLE LOWER SURFACE RADIATION EQUILIBRIUM TEMPERATURE

4.2 MATERIALS

4.2.1 Parawing Vehicle

The fuselage of the parawing vehicle, shown in Figure 7, can be broken down into three assemblies. They are the aft section and control surfaces, the propellant tanks, and the forward section.

The fuselage aft section supports the control surfaces and houses the rocket engines, recovery system, and control actuators. This section is constructed of Ti-6Al-4V titanium because of its superior strength-to-weight ratio at temperatures below 700°F as shown in Figure 20. Inconel alloy 718 (an age-hardenable nickel chromium alloy) was considered, but as average skin temperatures below 700°F were predicted, titanium was chosen. Structurally, the aft section is designed to withstand the thrust of all engines and the various loads induced by the control surfaces. An estimated average thickness of 1/4 inch of silica fiber insulation is provided.

The aft control surfaces are constructed of Inconel alloy 718 except for their leading edges and tips. These are fabricated from Hastelloy X (a nickel base alloy) because surface temperatures in excess of 1400°F are anticipated.

The propellant tank section consists of two cylindrical tanks end to end sharing one common bulkhead. The fuel and oxidizer are forced from their respective tanks by high pressure helium through outlets in the bottom aft part of each tank. Hard points are provided at either end of the tanks to bolt on the parawing. The propellant tank shell is welded from PH13-8Mo forgings (ellipsoidal ends and common bulkhead) and PH14-8Mo sheet (cylindrical sections). Selection of the PH13-8Mo tankage structure results in high strength, a high degree of toughness, and compatibility with both oxidizer and fuel. In addition, these materials are relatively easy to machine and weld and have good high temperature properties. The decision to use these materials was based on a comparison of high-strength alloys of aluminum, steel, and titanium. Titanium cannot be used because it is not compatible with the oxidizer. Aluminum tanks would be heavier than those fabricated of steel. The use of 17-4PH and 17-7PH in condition TH1100 is a possibility, but higher strength can be obtained with PH13-8Mo and PH14-8Mo. Maraging steel has attractive strength properties, but causes decomposition of hydrazine-type fuels.

The present tank design calls for an estimated average thickness of 1/4 inch of Armstrong Insulcork 2755 or equivalent. It is believed the 75 pounds of weight associated with the cork insulation can be reduced drastically by a low density insulation wrapped with a protective coating. However, this possibility was not investigated in sufficient detail for inclusion in this study.

No internal bladders, screens or other devices are contemplated which would retain propellants at the tank outlets during negative accelerations. In calm air, the flight path accelerations are such that a positive load factor is maintained at all times. At supersonic speeds, turbulent air is highly unlikely to result in negative accelerations, even instantaneously.

The forward section of the fuselage includes a titanium cylindrical shell, located between the nose cone and the tanks, which houses the battery pack,

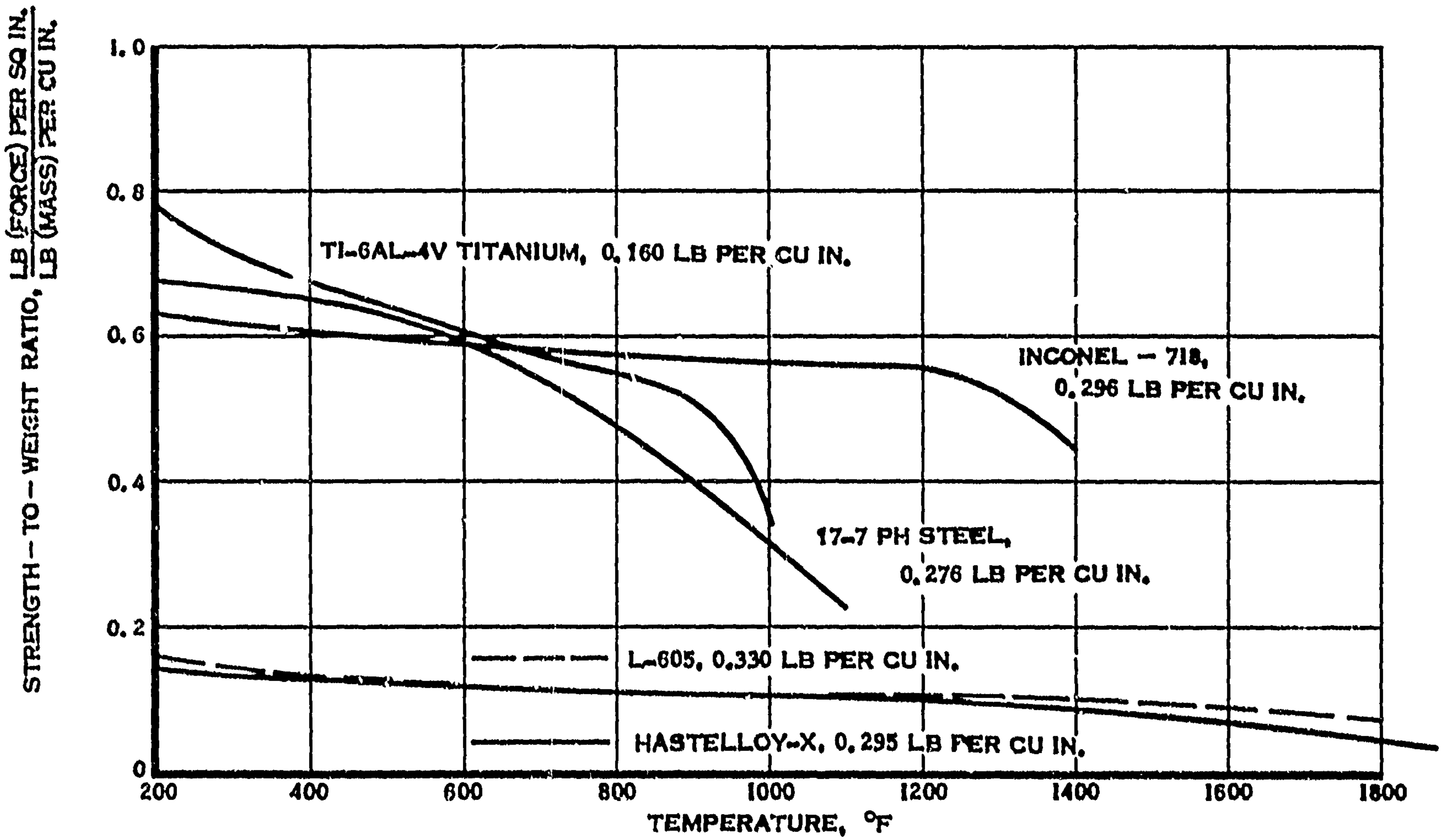


FIGURE 20. STRENGTH-TO-WEIGHT RATIO

and the pressurization and cooling system. The forward section also includes a nose cone of Inconel alloy 718 which houses the instrumentation, guidance, and control equipment. The forward section structure is designed to withstand loads induced by the air snatch during the recovery phase. The nose cone must withstand high aerodynamic loads during the boost phase along with temperatures that become very high near the tip. A Hastelloy X Q-ball (see Section 7) is mounted in the tip of the nose cone. For servicing and calibrating, the nose cone can be removed to expose the instrumentation, guidance, and control equipment which is mounted on a titanium beam. An estimated average thickness of 1/4 inch of silica fiber insulation is required.

Different size parawings would be installed depending on the planned altitude for cruise. The maximum area parawing is shown in Figures 4 and 8. The parawing consists of two leading edge booms which are elliptical sections of Ti-6Al-4V titanium, two spreader bars of L-605 alloy (an austenitic colbalt base alloy), a combined keel and deployment actuator of Ti-6Al-4V titanium, and a single-ply, metal fabric sail. The fabric is woven from fine wire of a nickel-chromium based alloy having a diameter of one mil or less. Two kinds of silicone rubber are used, one for impregnating and saturating the metal fabric and the other for an ablative heat shield. The ablative rubber can resist extremely high temperatures while maintaining the temperature at the interface between the char and parent elastomer at 1100°F (Reference 3).

The sail material weighs 0.264 pounds per square foot and has a yield strength of 278 pounds per foot. The leading edge booms are designed to withstand bending loads about the spreader bars and about the hinge pins under a 3g sail load. These booms are thermally protected by the sail material wrapped around their leading edges. The spreader bars are constructed of L-605 alloy because of their small diameter and low sweep angle which cause the stagnation temperature to exceed 1400°F. They hold the leading edge booms in the deployed attitude and carry primarily a column compressive load. The parawing keel is designed to resolve the bending loads at the hinge pins into loads at the fuselage hard points, and to secure the sail at the centerline of the parawing. It also serves as the cylinder for the parawing deployment actuator. The deployment actuator is operated by high pressure helium from the fuel expulsion system. The actuator push rod moves a slide which is attached to the spreader bars, thereby deploying the parawing.

4.2.2 Lifting Body Vehicle

The second vehicle that was studied was a lifting body configuration as presented in Figure 3. This configuration has its lower surface, forward upper surface, vertical fins, and control surfaces constructed of Inconel alloy 718. This alloy is utilized because of surface temperatures in excess of 700°F in these areas. The aft upper surface, the inner skin and the internal structure, with the exception of the oxidizer tank, are constructed of Ti-6Al-4V titanium. In these areas the expected temperatures are below 700°F and titanium has a better strength-to-weight ratio. The vehicle is insulated by an estimated average thickness of 1/4 inch silica fiber insulation bonded to the inner surface of the outer skin.

The instrumentation, guidance and control equipment is located in the forward part of the vehicle. The individual components are mounted on a titanium keel for mechanical support. The nose can be removed intact, thereby exposing the equipment for easy maintenance and checkout. A Hastelloy X Q-ball is mounted in the tip of the nose cone.

The control surfaces are constructed of Inconel alloy 718 skins and ribs. The vertical fins are fixed to the aft sides of the vehicle. They have surface skins of Inconel alloy 718 and a honeycomb core.

SECTION 5

SYSTEM PERFORMANCE ANALYSIS

5.1 AERODYNAMICS

Supersonic force data are presented in Figures 21 to 28 for both the lifting body and the parawing vehicles. The data for the parawing is split into fuselage, wing and tail components because any size parawing may be installed up to a certain maximum size and also because of the different trim angle requirements for the horizontal tail. Interference drag was small enough to be neglected. Some of the data are presented in terms of the normal force coefficient, C_N, and the drag at zero lift, C_{D_o}. The lift and drag coefficient, C_L and C_D, are easily derived as

$$C_L = C_N \, \text{Cos}\alpha$$

$$C_D = C_{D_o} + C_N \, \text{Sin}\alpha$$

where α is the angle of attack.

The normal force characteristics of the lifting body vehicle were obtained from Reference 5 by assuming that the vehicle could be simulated by a delta wing of the same planform and dimensions. The drag at zero lift was taken from wind tunnel test data.

The fuselage normal force data for the parawing vehicle were taken from Reference 6 while the tail lift and drag data were taken from Reference 5. Parawing characteristics were taken from Reference 7 assuming a canopy inflation angle of 100 degrees. Since the data in that reference was test data taken at Mach 6.6 only, it was modified for Mach number by assuming that, at a particular angle of attack, the test values could be multiplied by the ratio of lift for a flat plate at Mach numbers of 1.5, 2, 4, or 6 and lift at the reference Mach number of 6.6.

In order to find the tail lift, it was necessary to determine the tail angle required to balance out the aerodynamic moments about the vehicle center of gravity. The necessary moment data for the fuselage was taken from Reference 6. The required tail angles presented in Figure 29 were calculated for the no wing case. At intermediate altitudes where a small wing will suffice, the wing can be placed so as to maintain the same relation between the angle of attack of the fuselage and the angle of attack of the horizontal tail. However, when the largest wing is installed, the same relationship cannot be maintained. For this configuration, it was found that the following formulas for the lift, L, and the lift-to-drag ratio, L/D, could be used with sufficient accuracy for the performance calculations:

$$L = 1.3 L_{wing}$$

$$L/D = (L/D)_{wing}$$

The aerodynamic data takes the usual form expected as the speed increases in the supersonic regime: The lift curve slope and the lift-to-drag ratio fall

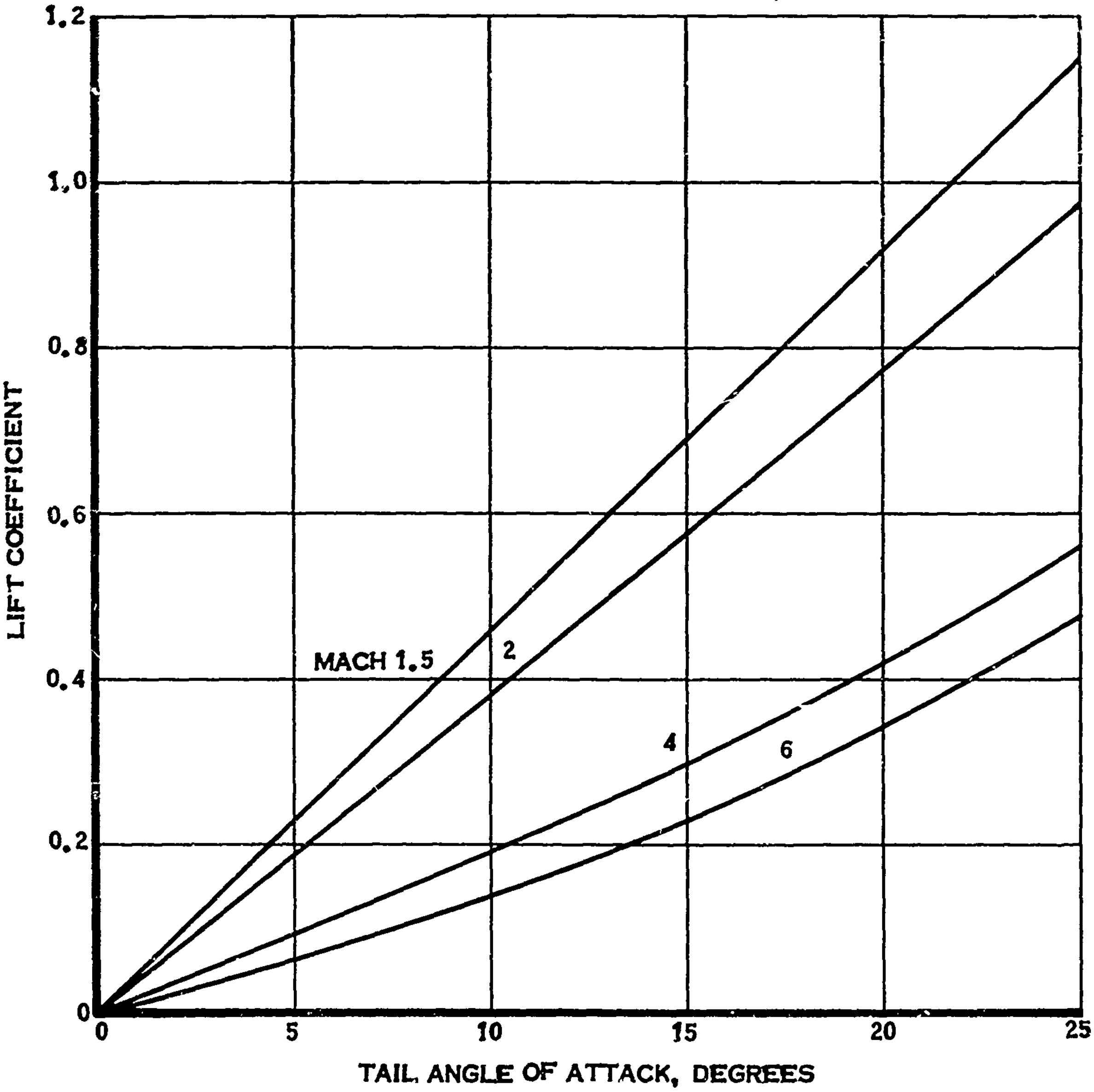


FIGURE 21. PARAWING VEHICLE TAIL LIFT

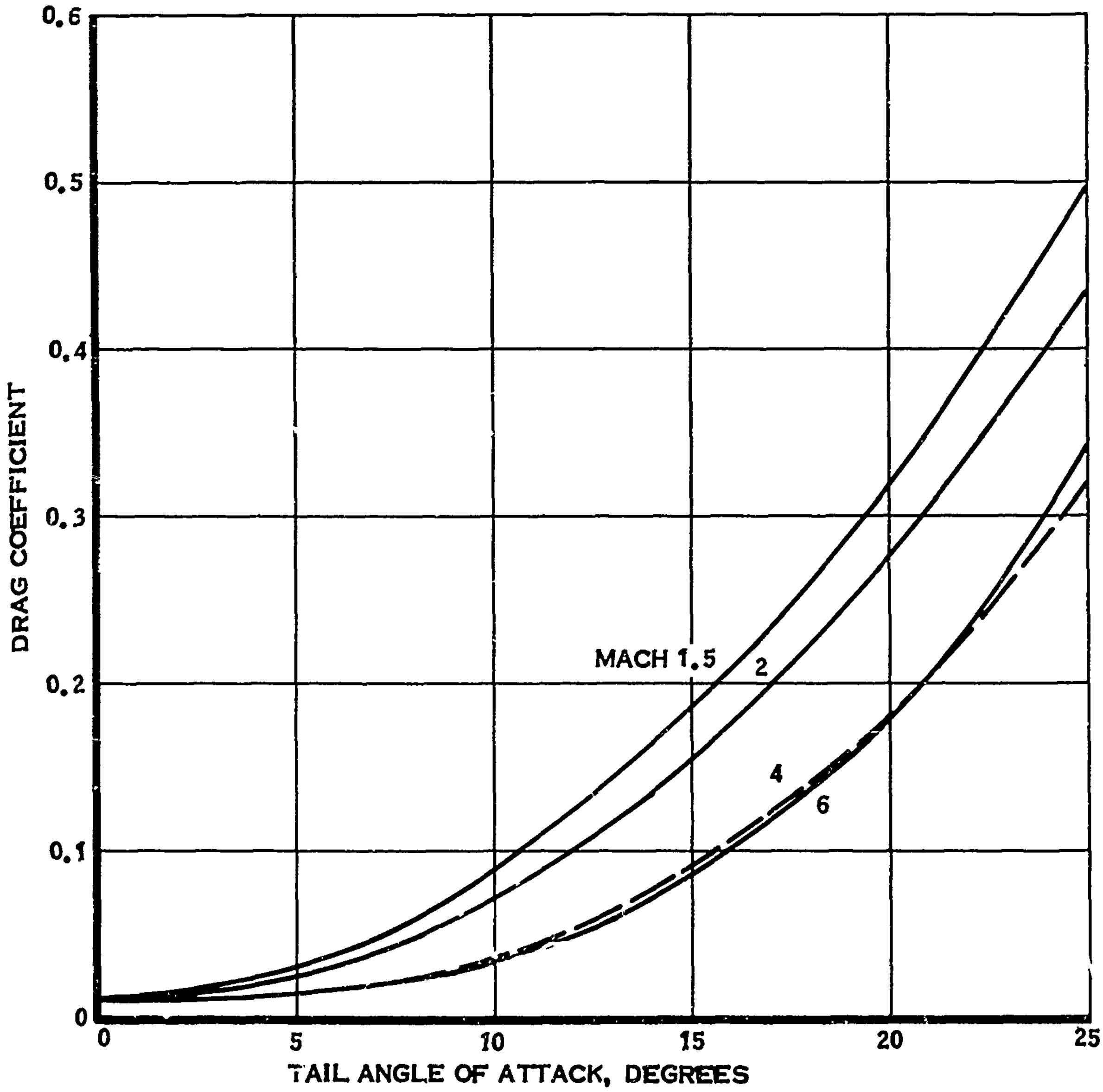


FIGURE 22. PARAWING VEHICLE TAIL DRAG

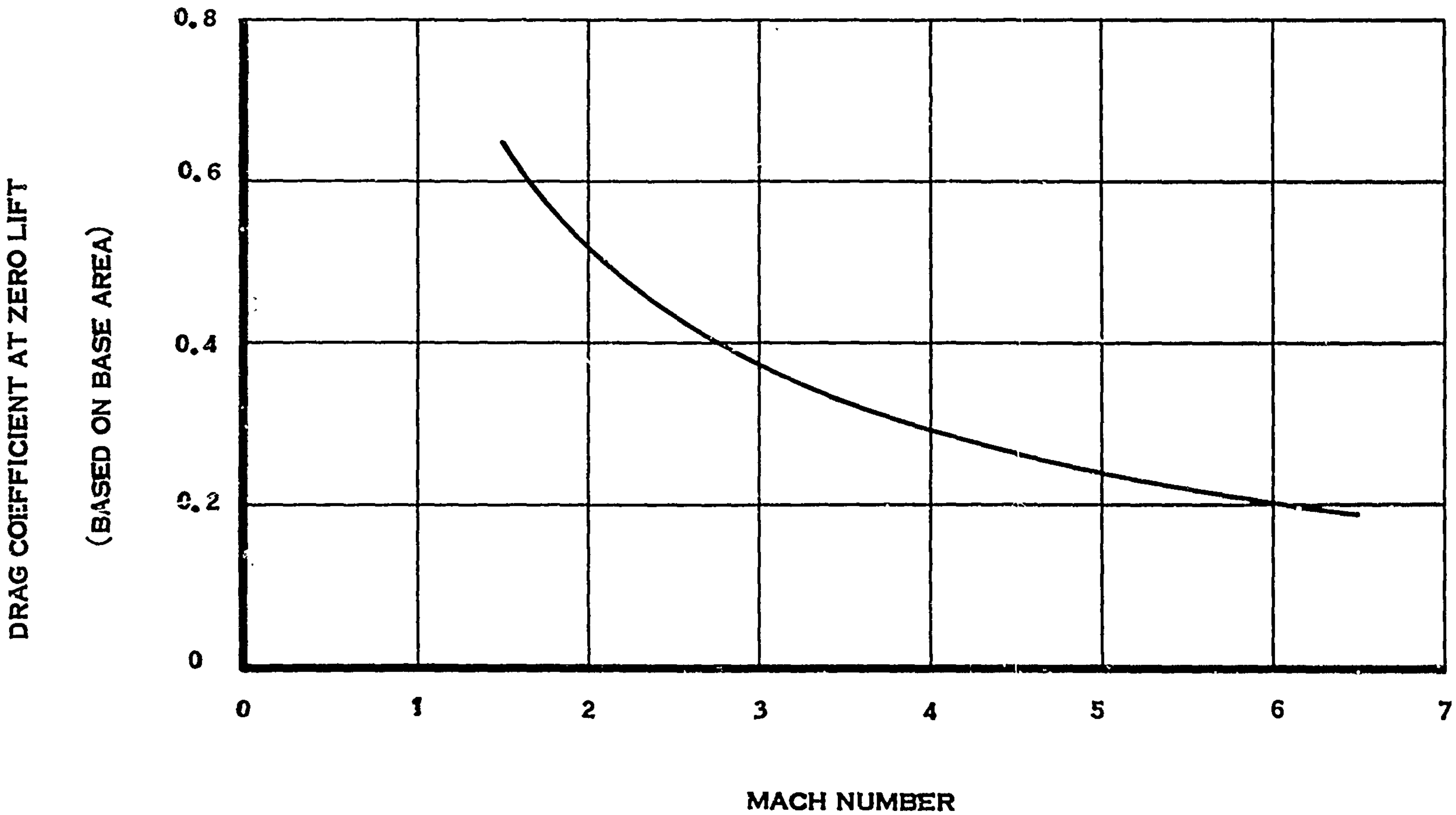


FIGURE 23. PARAWING VEHICLE FUSELAGE DRAG AT ZERO LIFT

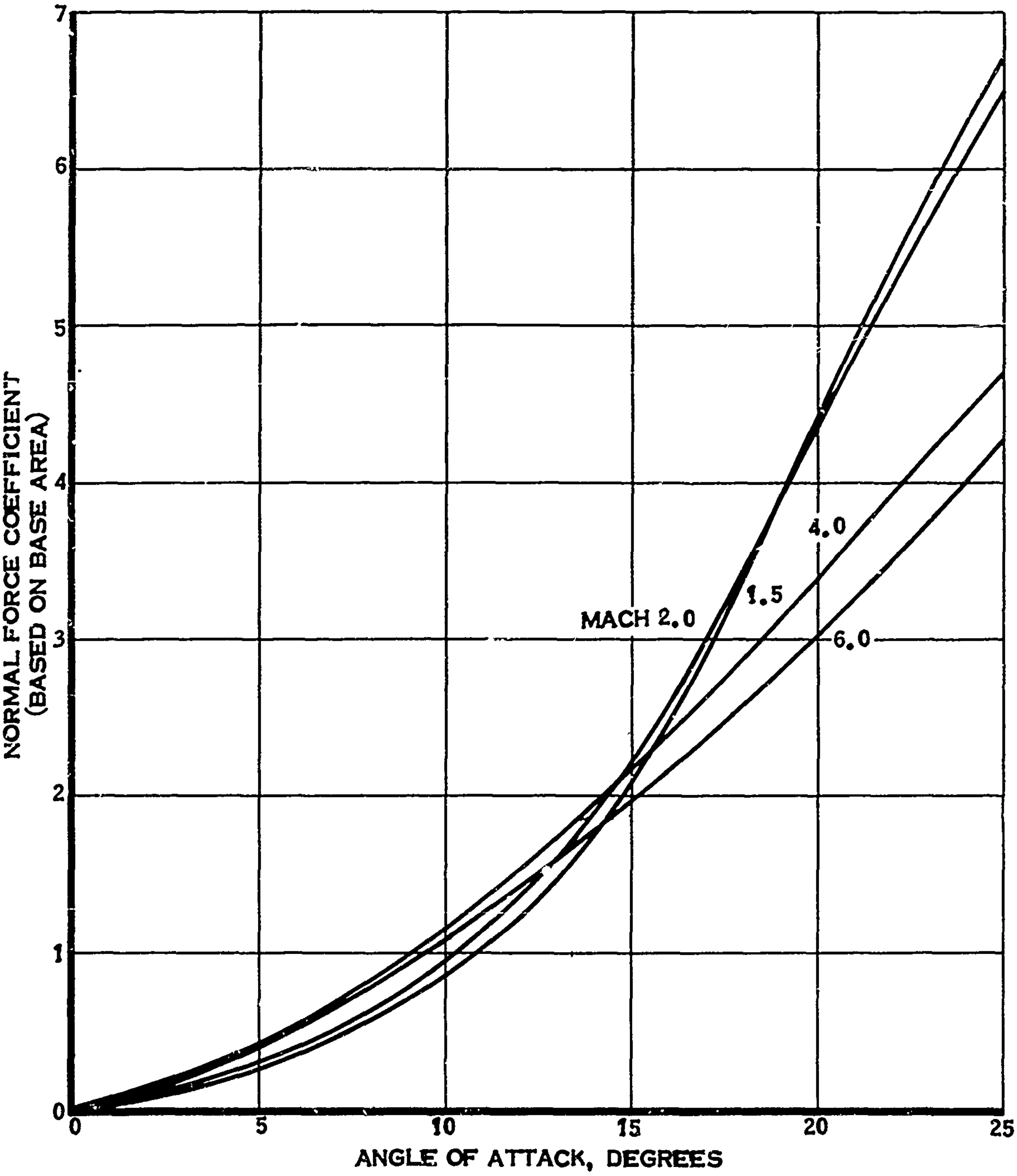


FIGURE 24. PARAWING VEHICLE FUSELAGE NORMAL FORCE

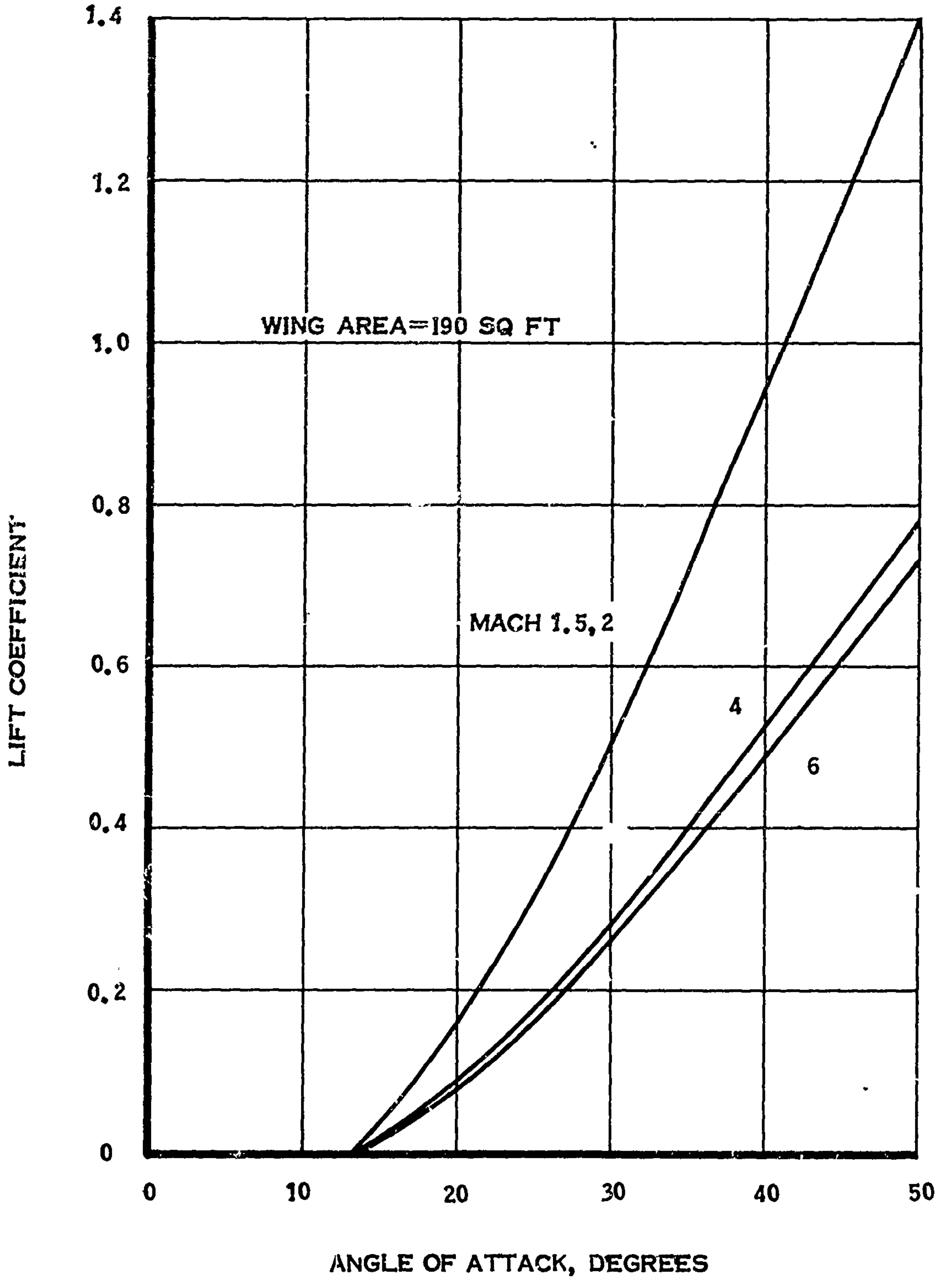


FIGURE 25. PARAWING VEHICLE WING LIFT

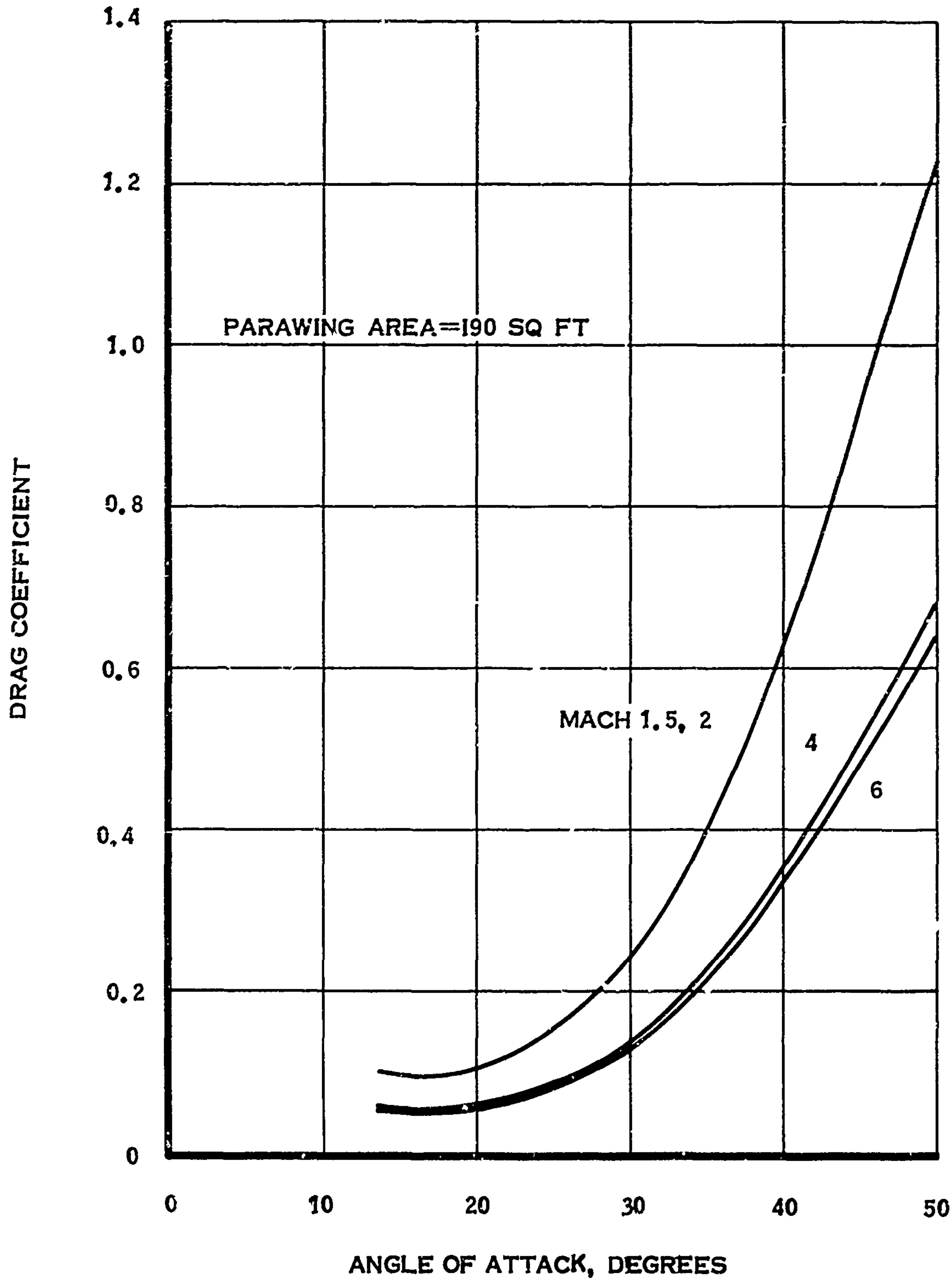


FIGURE 26. PARAWING VEHICLE WING DRAG

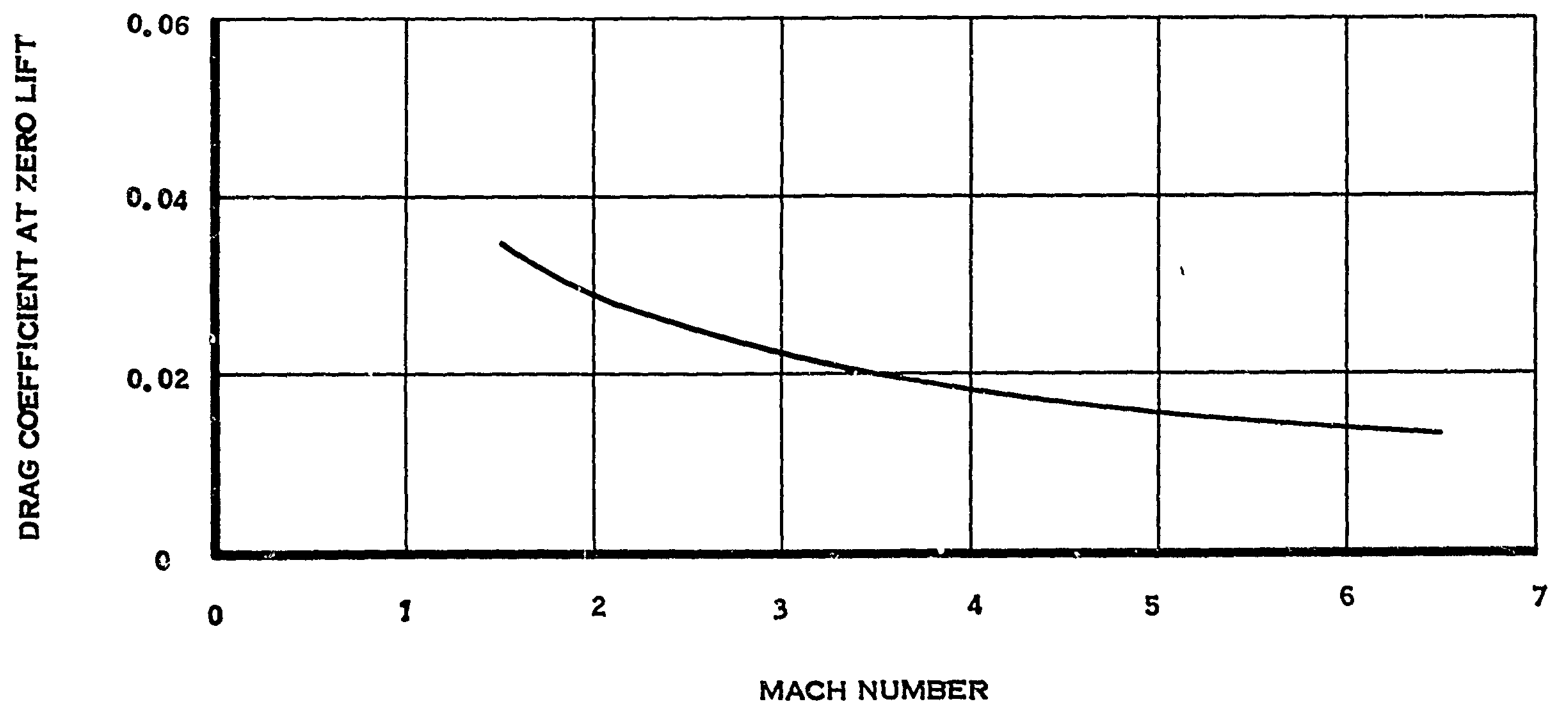


FIGURE 27. LIFTING BODY VEHICLE DRAG AT ZERO LIFT

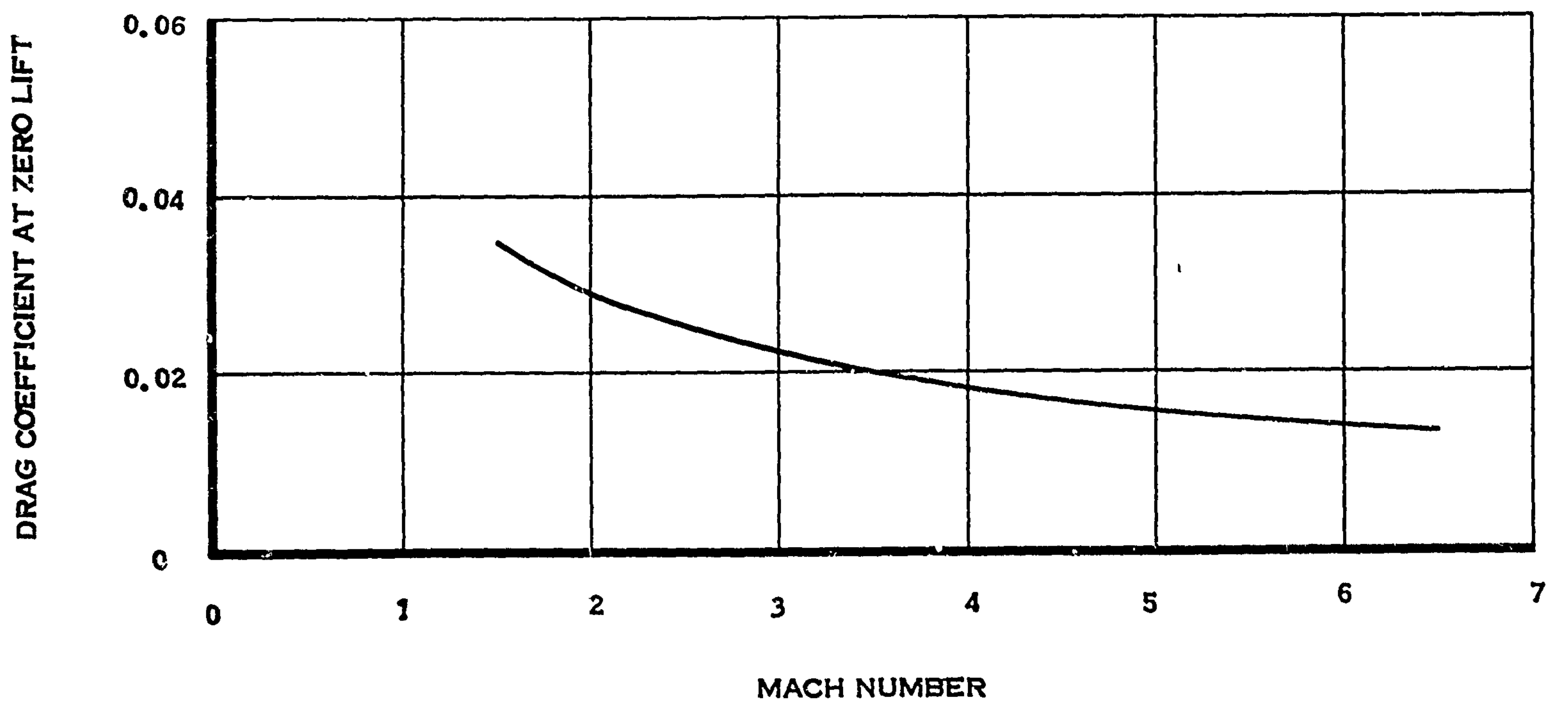


FIGURE 27. LIFTING BODY VEHICLE DRAG AT ZERO LIFT

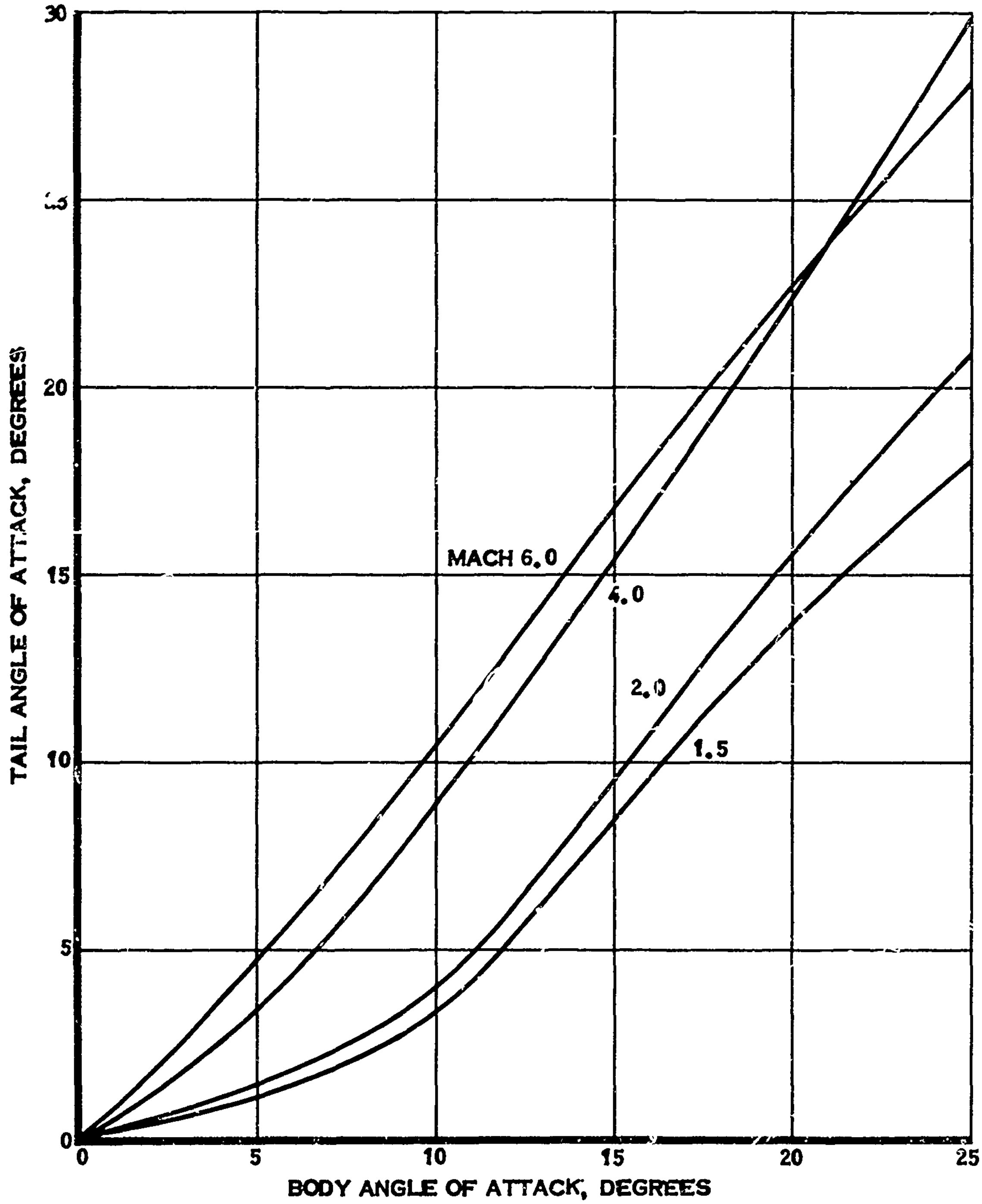


FIGURE 29. PARAWING VEHICLE WITHOUT WING, TAIL ANGLE OF ATTACK

off generally with Mach number. The resulting lift-to-drag ratios, although reasonable for the parawing vehicle with or without a wing, appear to be somewhat low for the lifting body, being only slightly above 3 at Mach 4 and 6. The cross flow assumption used herein for calculating the drag may be pessimistic, especially at the lower speeds. There is, however, no test data for lifting bodies at the speeds of interest and the present data is applicable only to hypersonic speeds. A development program for a lifting body vehicle should provide for detailed aerodynamic analysis and wind tunnel tests on variations of the lifting body shape to achieve higher lift-to-drag ratios.

5.2 PERFORMANCE

5.2.1 Boost Performance

Boost trajectories were calculated for both the parawing and the lifting body vehicles for a variety of initial weights, final weights and launch angles. A Lockheed computer program was available, Reference 8, which was readily adaptable to the study. It incorporates the normal trajectory equations for a point mass and includes refinements such as allowing for a spherical earth. The program adjusted the rocket thrust level with altitude so that it approximated the variation indicated in Figures 10 and 12 (for eight thrust chambers). The drag was derived from the expression $C_{D_o} Sq$ where C_{D_o} is the drag coefficient at zero lift, S is the reference area and q is the dynamic pressure. Small variations in C_{D_o} and S would result from the installation of different sizes of parawings which would be folded back along the top of the fuselage during boost. This variation has only a small effect on boost performance and was not accounted for.

The launch maneuver is a pullup from level flight to the required launch angle and always begins at the same entry speed and altitude. An entry altitude of 35,000 feet was chosen since the launch aircraft, an F-4C, is known to achieve its best supersonic acceleration rate with a 600 gallon tank as a centerline store at altitudes near 35,000 feet. The selection of the entry speed was also based on the supersonic acceleration of the F-4C aircraft with a 600 gallon tank. Higher speeds are possible, but the chosen velocity of 1900 fps is believed to be about the highest practical.

Speed will be lost and altitude gained during the pullup to the launch attitude angle as shown in Figure 30. The loss in velocity during the pullup to launch is taken to have a value somewhat less than that for an F-104 aircraft launching two large rockets from its wing tips. Less velocity loss exists for the F-4C aircraft because the F-4C has more excess thrust than the F-104. The F-104 data was obtained from computer trajectories and is available in Reference 9. The load factor during the pullup would be 3g, the maximum that the HI-HICAT vehicle is designed for. The altitude gain was also taken from the F-104 data but no allowance was made for the higher excess thrust available for the F-4C since the difference is minimal.

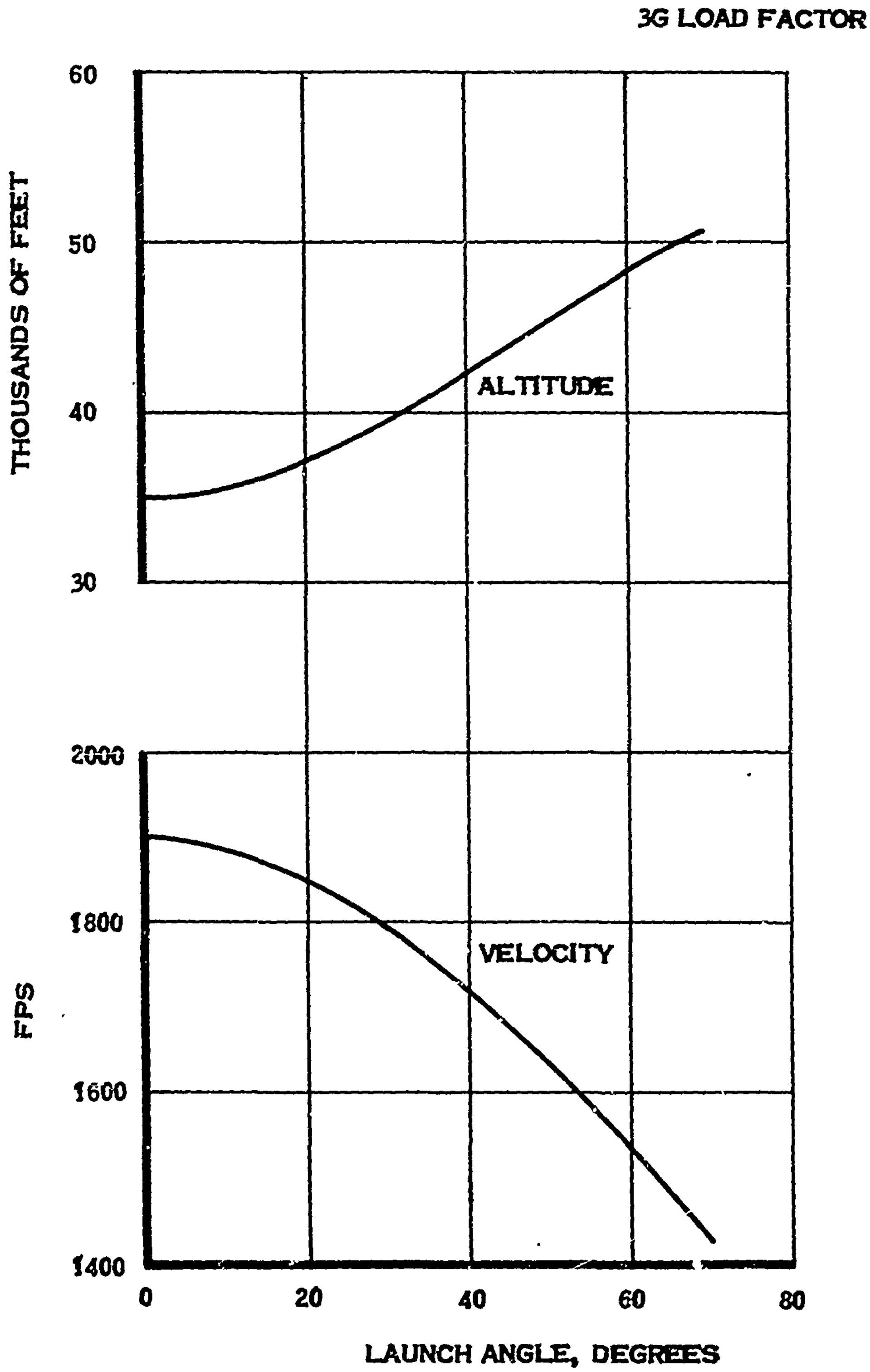


FIGURE 30. LAUNCH MANEUVER

Figure 31 presents a typical time history of a high speed, high altitude trajectory. These trajectories were calculated for the case where the sustainer engine is shut down until the cruise altitude is reached. However, the propellant feed system can be simplified if the sustainer is not shut off and a positive longitudinal acceleration is maintained. The difference in boost performance would be minimal.

Plots summarizing all the boost trajectory data are presented in Figures 32 and 33. The mass ratio given in the figures is the ratio of the weights at the beginning and end of boost. Note that the lifting body vehicle takes slightly more propellants to reach cruise speed, because both weight and drag are slightly higher.

Calculations were completed for both eight and nine thrust chambers for the parawing vehicle. These computations indicated the increase in performance resulting from the addition of another chamber was small and that eight was near optimum. Calculations were also completed for two different initial weights, 2507 and 2724 pounds, again for the parawing vehicle only. The difference in performance was too small to be plotted in Figure 32.

5.2.2 Range

The cruise range was computed from the formula

$$R = I\ V \left[\frac{L}{D} \ (\mathrm{Cos}\ \alpha) + \mathrm{Sin}\ \alpha \right] \mathrm{Log}\ (W_o / W_1)_{cruise}$$

where R is the range, I is the specific impulse, V the cruise velocity, a constant, and W_o and W_1 are the vehicle weights at the start and end of cruise, respectively.

Since each calculation was completed at a constant angle of attack, a cruise climb condition results in which the altitude increases as the weight of onboard propellants drops. The results of these calculations are presented in Figures 34 to 36 where the altitudes indicated are values obtained at about the mid point of the cruise. The mass ratio in the figures is defined as the ratio of the vehicle filled with propellants at the start of the boost to the weight empty. Hence these plots have taken into consideration the propellants used during the boost.

Range was calculated for only the no wing case and the case where the largest parawing is installed as plotted in Figures 34 and 35. The parawing vehicle with intermediate sizes of parawing will achieve ranges greater than those presently plotted at altitudes between 130,000 and 170,000 feet.

The range plots indicate higher cruise speeds should be selected at the higher altitudes. Part of the increased range at higher speeds is due to the considerable increase in range possible by allowing the HI-HICAT vehicle to decelerate at altitude to Mach 1.5 or to an angle of attack of 50 degrees. The maximum attitude angle was selected arbitrarily but the Mach 1.5 condition exists because the recovery requirements and the desire not to fly at transonic or subsonic speeds.

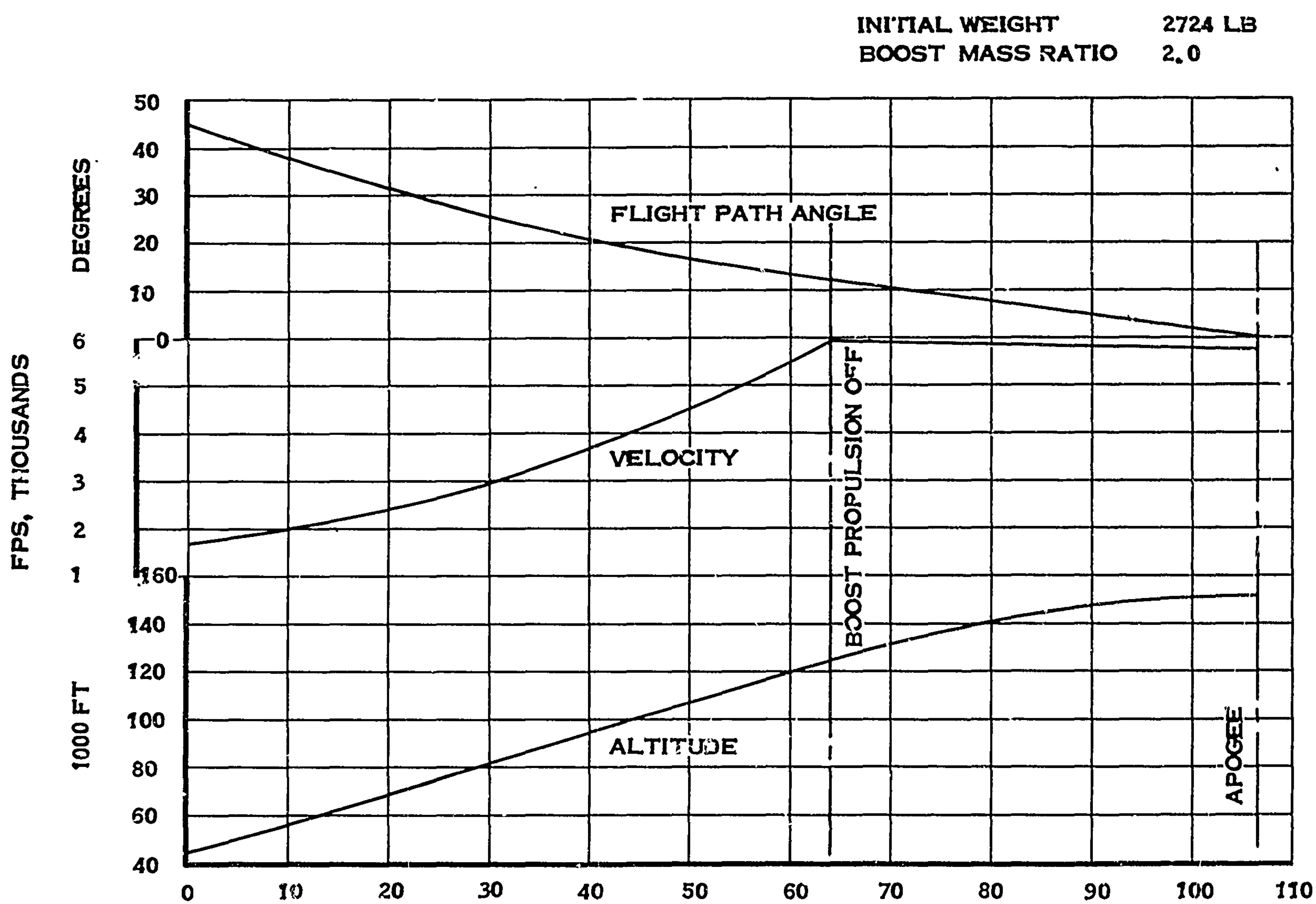


FIGURE 31. PARAWING VEHICLE BOOST TRAJECTORY

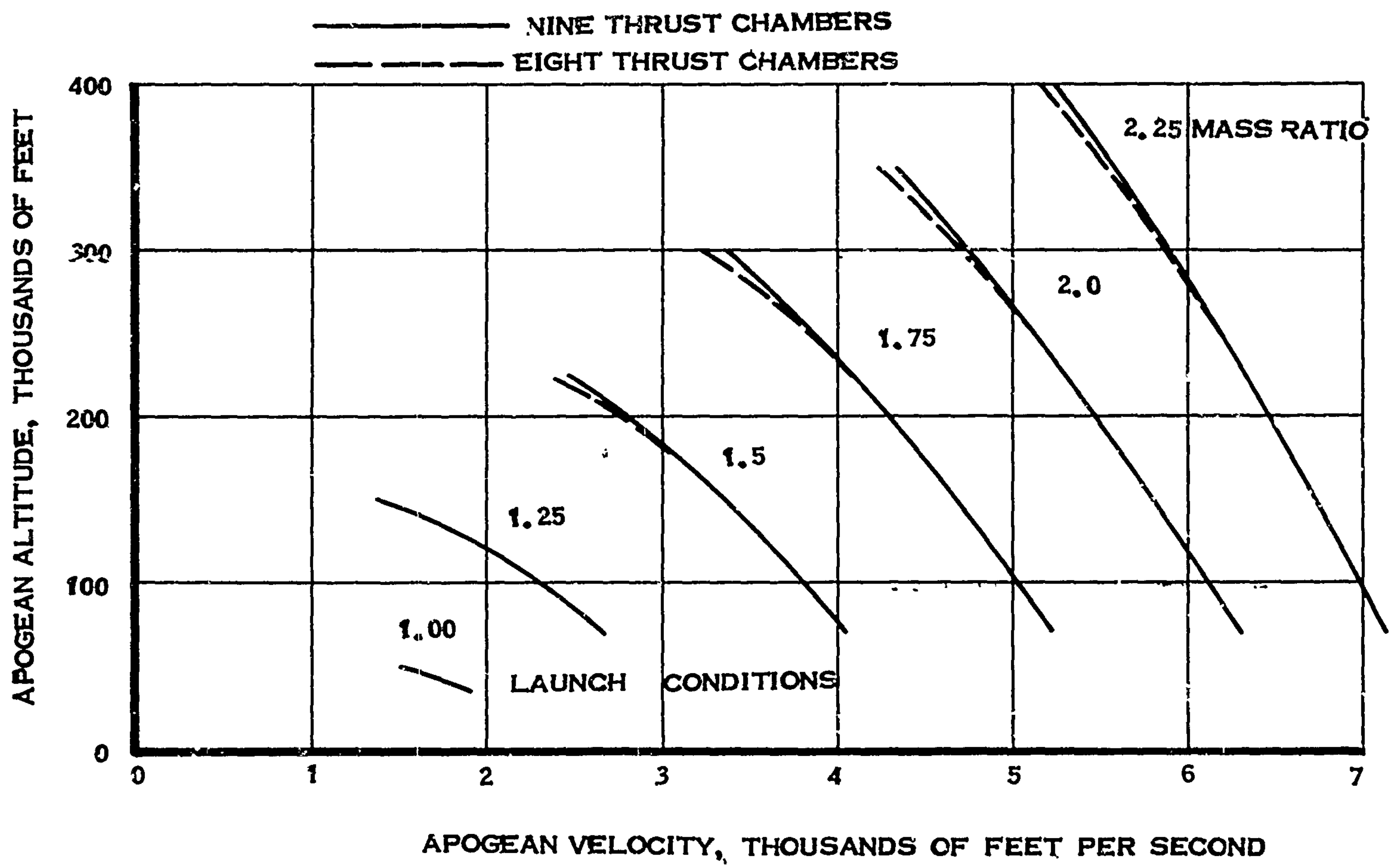


FIGURE 32. PARAWING VEHICLE BOOST PERFORMANCE

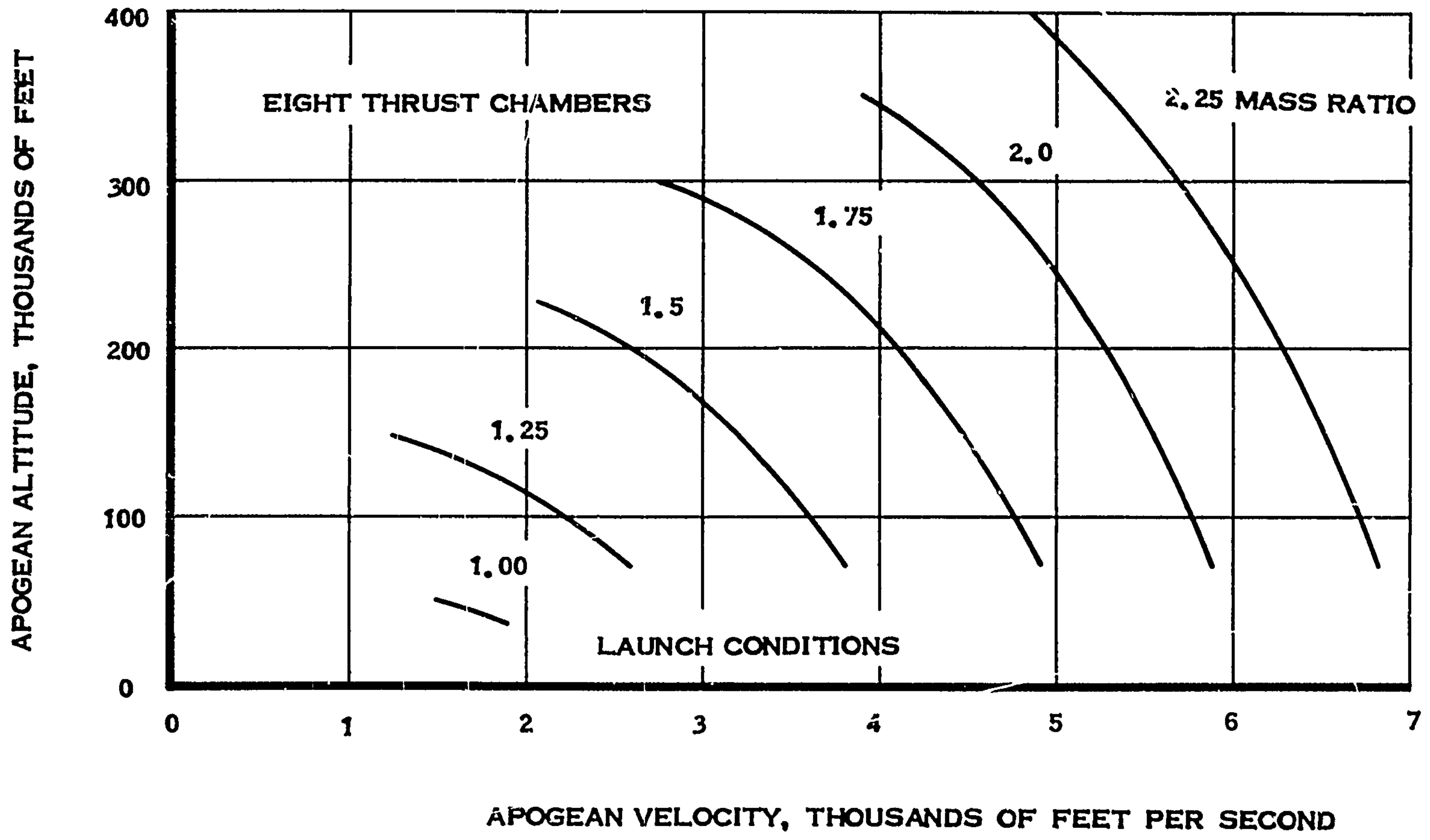


FIGURE 33. LIFTING BODY VEHICLE BOOST PERFORMANCE

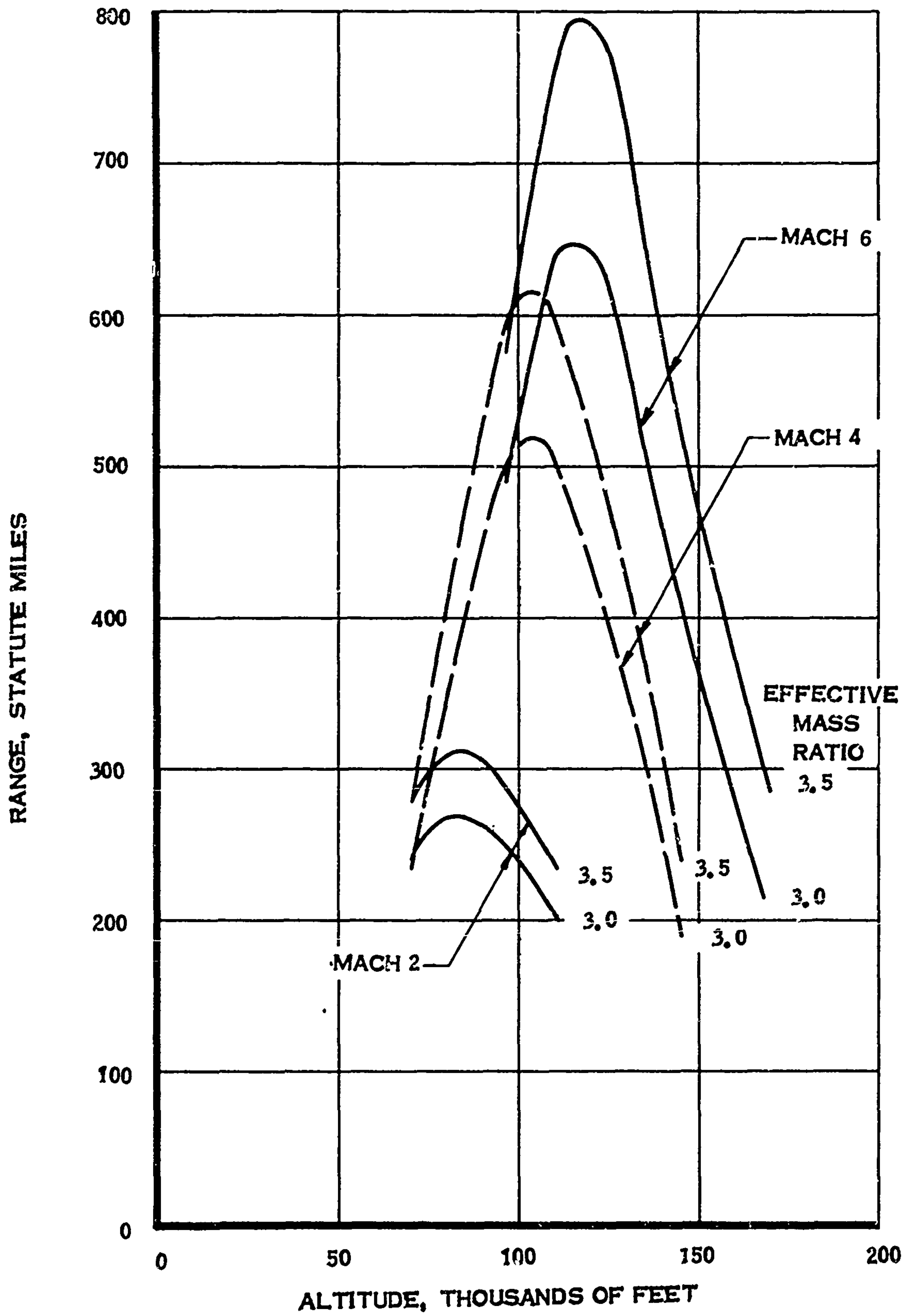


FIGURE 34. PARAWING VEHICLE WITHOUT WING, RANGE DURING CRUISE AND DECELERATION

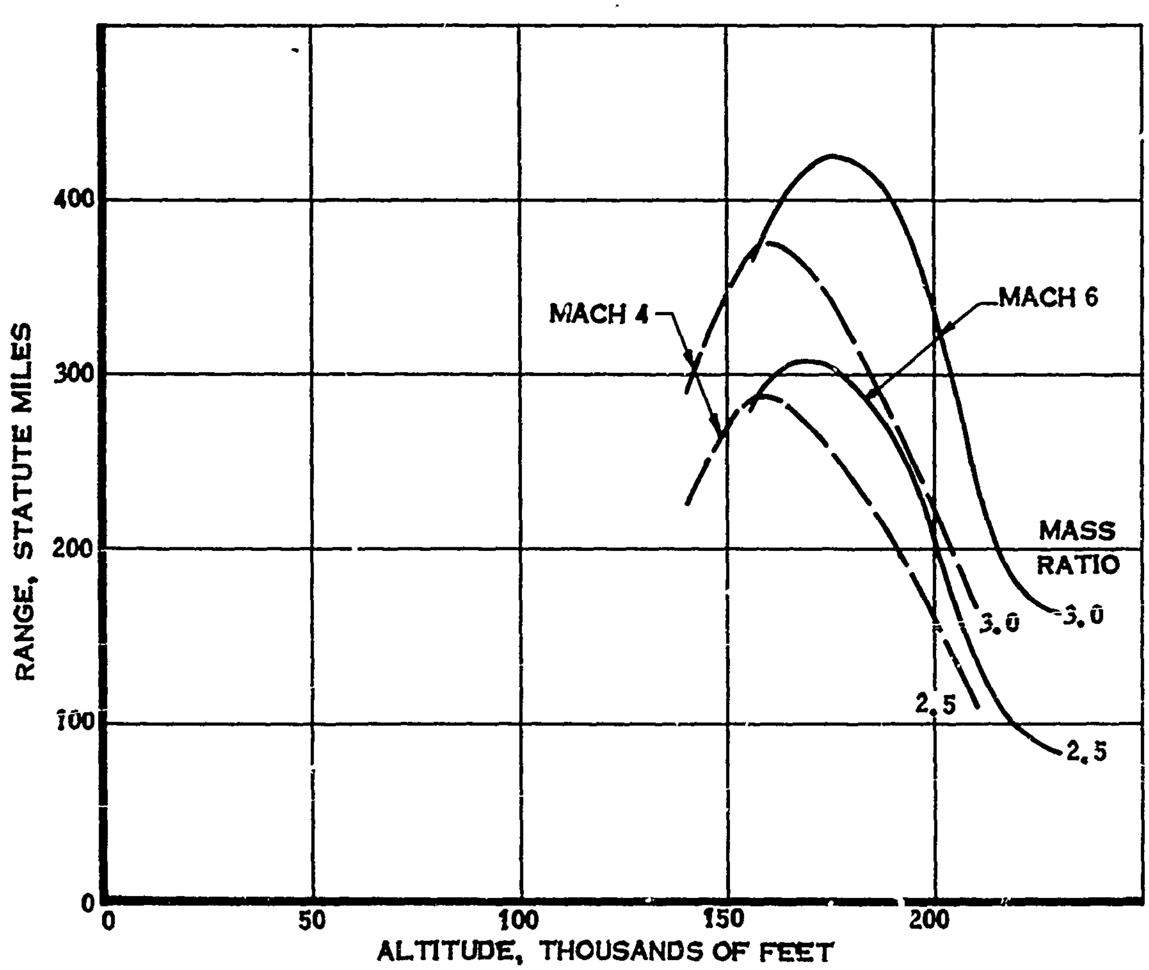


FIGURE 35. PARAWING VEHICLE WITH LARGE PARAWING, RANGE DURING CRUISE AND DECELERATION

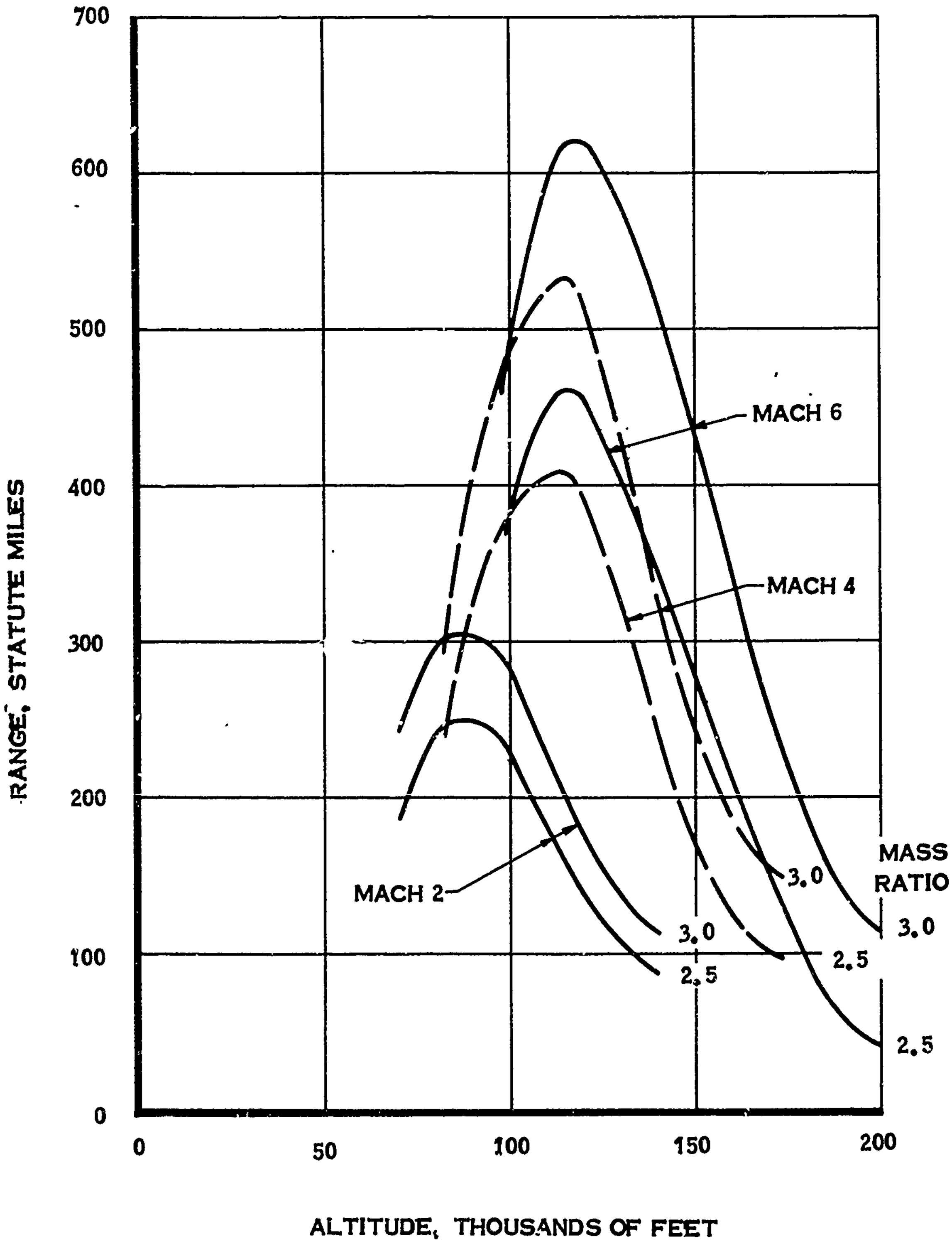


FIGURE 36. LIFTING BODY VEHICLE, RANGE DURING CRUISE AND DECELERATION

The range available during deceleration is indicated in Figure 37 for the lifting body vehicle. This range is important at higher speeds since it is increasing as the velocity squared as indicated by the relation

$$dR = V\ dt = V \frac{dV}{a} = 1/2 \frac{dV^2}{a}$$

where

$$a = \frac{D}{L} g$$

and where a is the longitudinal deceleration and g the acceleration of gravity.

The weight statement of Section 9 indicates mass ratios of 2.77 and 3.24 for the parawing vehicle with and without the large parawing. The mass ratio is defined as $(W_o / W_1)_{overall}$

where

$$W_o = W_p + W_h + W_1.$$

W_p and W_h are the weights of the propellant and helium, and W_o and W_1 are the full and empty weights, respectively. Unfortunately, in deriving the mass ratio for the lifting body vehicle an allowance must be made for the propellants consumed by the turbopump. The parawing vehicle uses a highly pressurized tank and does not suffer this particular decrement in available propellants. This consumption of propellants averages about 6 per cent (125 pounds of propellants) of the total propellants used, making the effective mass ratio of the lifting body vehicle about 2.7.

Using the above values for the mass ratio in the range plots would lead to the conclusion that greater range can be achieved with the parawing vehicle. It should be emphasized, however, that the aerodynamic characteristics of the lifting body vehicle were not optimized and that, indeed, superior versions of the vehicle are being continually developed.

5.3 STABILITY

The longitudinal and lateral stability of the lifting body configuration that was used for this study has been demonstrated by test data. This configuration has static stability and requires trim type actuators only.

The longitudinal stability of the parawing vehicle has been investigated in a preliminary manner by analyzing the stability of the vehicle without wings and with the maximum wing area. The method of Reference 10 indicates that at low angles of attack the wingless vehicle will be longitudinally stable at all the Mach numbers of interest. At higher angles, the stability was investigated by increasing the vehicle angle of attack from the trimmed condition while holding the angle between the tail and body fixed, then determining whether the resulting force on the tail and body was such as to bring the vehicle back to the original angle of attack. The results of the analysis indicate that although the vehicle has longitudinal static stability with the maximum wing

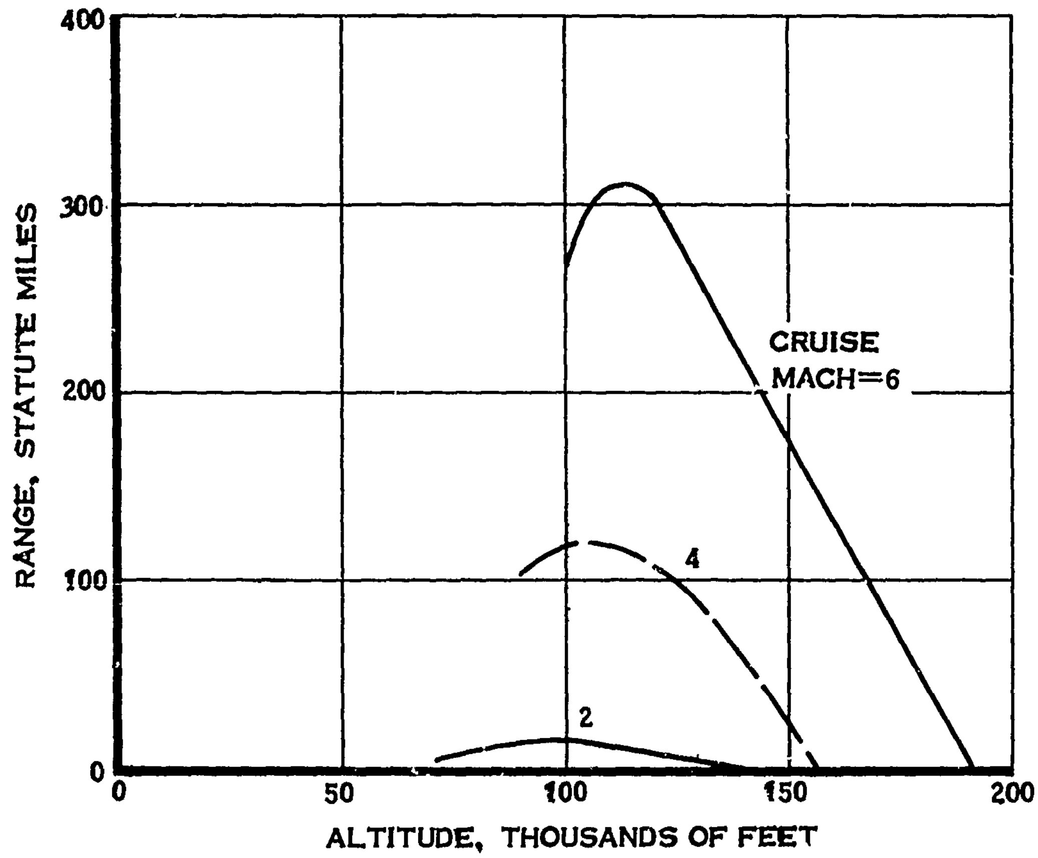


FIGURE 37. LIFTING BODY VEHICLE RANGE DURING DECELERATION

area, it will be statically unstable at high angles of attack, exceeding 10 degrees, if flown without wings. This instability can be corrected by increasing the size of the horizontal tail fins. However, it should be noted that even when the vehicle is unstable it is still controllable (with an autopilot) since the horizontal tail angles required from trim are not excessive. This situation is not uncommon in missile designs.

The lateral stability of the wingless parawing vehicle has been investigated using the tail data of Reference 10 and the centers of pressure data for the sideslip forces, Reference 5. It has been established that the wingless vehicle is stable to lateral deflections. Since the center of pressure (for lateral forces) of the wings will be behind the vehicle center of gravity, the winged vehicle should also be laterally stable.

Roll stability of the winged vehicle should not present any problems, as the moment due to the vertical tail will be balanced by the moment due to the parawing. However, if the vehicle is flown without a wing, the fact that the vertical tail is on the bottom tends to make this version of the vehicle inherently unstable during rolls. Since the entire area of both horizontal tails can be moved separately, this roll instability can be controlled by pitching the horizontal tails in such a manner that they provide a moment that balances the roll moment due to the vertical tail.

SECTION 6

COMMAND AND CONTROL

6.1 DESIGN FACTORS

The guidance and control system must stabilize the vehicle and navigate it to a predetermined location within range of the recovery aircraft. Command guidance systems, as presently used in target drones, depend upon radar as a feedback link which detects the error between the vehicle's position and the desired position so that the command operator can transmit signals to the vehicle's remote control system and control its speed, altitude, heading or attitude. Radar, however, cannot provide data of sufficient accuracy to measure vehicle motions due to turbulence of 1 fps as required for this study at the high frequencies of interest (of the order of 10 cps). This failing is due to atmospheric turbulence and other difficulties which limit accuracies to a few parts per million (Reference 11) for even the most sophisticated of triangulation radars. Such an argument indicates the desirability of having a self-contained, onboard guidance system since some of the required turbulence instrumentation can then be shared with the guidance equipment.

An inertial navigation system will be shown to be practical in this study. It is a system which can be completely independent of intelligence radiated from the ground. However, it is deemed wise to provide for ground control commands to override those from the inertial equipment in the interest of increased reliability. Admittedly, additional costs will be incurred by requiring radar and/or telemetering receiving stations, but it is better to provide for a function which requires relatively little vehicle equipment than to try to add this function at a later date.

An inertial navigation system is self-contained and all the data necessary for navigation and control of the vehicle in flight is calculated and programmed into the system's computer prior to launch. The inertial navigation system can then determine deviations in heading and attitude, and automatically operate the control surfaces to stabilize the vehicle and correct its course.

Since precision computing is not necessary for navigation or in-flight control, it is not necessary to install a large computer in the vehicle. A small computer and inertial platform with precision sensors to provide data for telemetering and guidance is all that is necessary. Such small inertial systems called LCI (Low Cost Inertial Navigation Systems) have been developed recently by a number of companies such as Teledyne, Lear, and Kearfott.

These inertial navigating systems are small and lightweight. The Teledyne system which was selected for this study weighs only ten pounds. It consists of three units; an inertial sensor, a small reference computer, and a power supply in a single package as illustrated in Figure 38. The system characteristics are listed in Table 3. A small inertial platform is the heart of the system and weighs only 5 pounds. The stable element consists of gyros with two degrees of freedom and three accelerometers mounted in a four-gimbal arrangement.

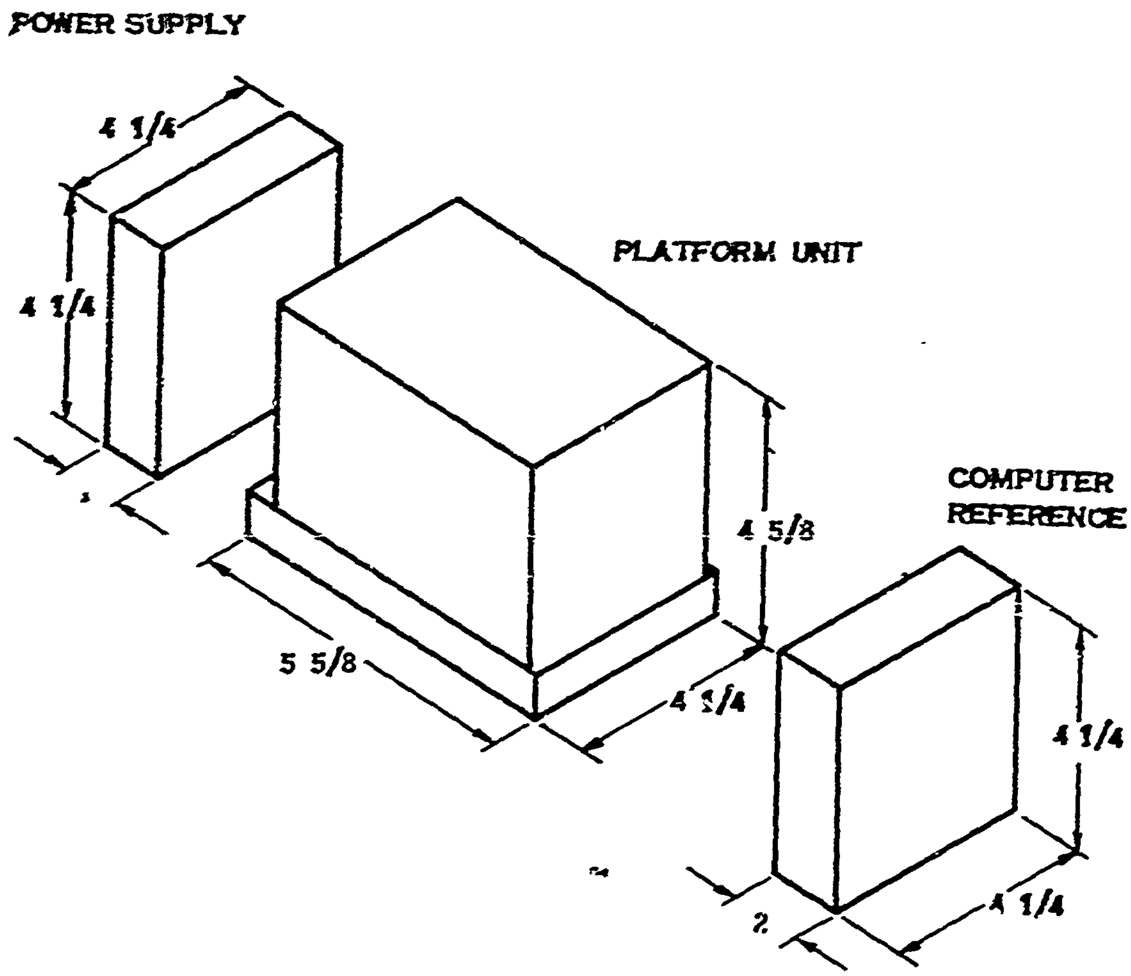


FIGURE 38. INERTIAL DATA SYSTEM

TABLE 3

INERTIAL DATA SYSTEM CHARACTERISTICS

Physical		**Alignment**	
Weight	10 Lb	Fine Align Accuracy Level	0.006 Deg
Operation Time Without Power	10 Min	Azimuth (repeatability of reference)	0.0085 Deg
Cooling	Self-contained	Fine Align Time (from turn-on at 70°F)	1.5 Min
Number of Units	3	Gyrocompass Accuracy	0.15 Deg
System Outputs		Gyrocompass Time (from turn-on at 70°F)	6 Min
Acceleration	Pulse Rate	Warmup Time	1 Min at 60°F
Velocity Components	Pulse Rate	**Performance Accuracy**	
Present Position (Visual)	Latitude-Longitude		
Range Bearing	Digital	Verticality	0.025 Deg
Attitude	Synchro	Position	3 Nautical Miles per Hr
Heading, Free	Synchro	Heading	0.08 Deg per Hr
Electrical		Vertical Velocity	10 fpm per Min
Voltage	28 Volts dc	Mean Time Before Failure	1000 Hours
Power	25 Watts		
Environmental Power, +28 V	56 Watts		

At altitudes approach[illegible]0 feet, the atmosphere is too thin to make pressure altitude mea[illegible] with barometric altimeters. By integrating the outputs of the platf[illegible]rtical accelerometer as shown in Figure 39, the altitude can be calc[illegible]. Integrators used for this purpose are subject to drift and produce an altitude error which increases with time. Usually an external reference such as a barometric altimeter is used to damp the altitude integrator output. The integrator error is limited by the short duration of the HI-HICAT missions, and if a barometric altimeter is used to damp the integrator signal up to 70,000 feet, the altitude drift error is limited to approximately 5000 feet rms at 200,000 feet altitude.

6.2 GUIDANCE SYSTEM

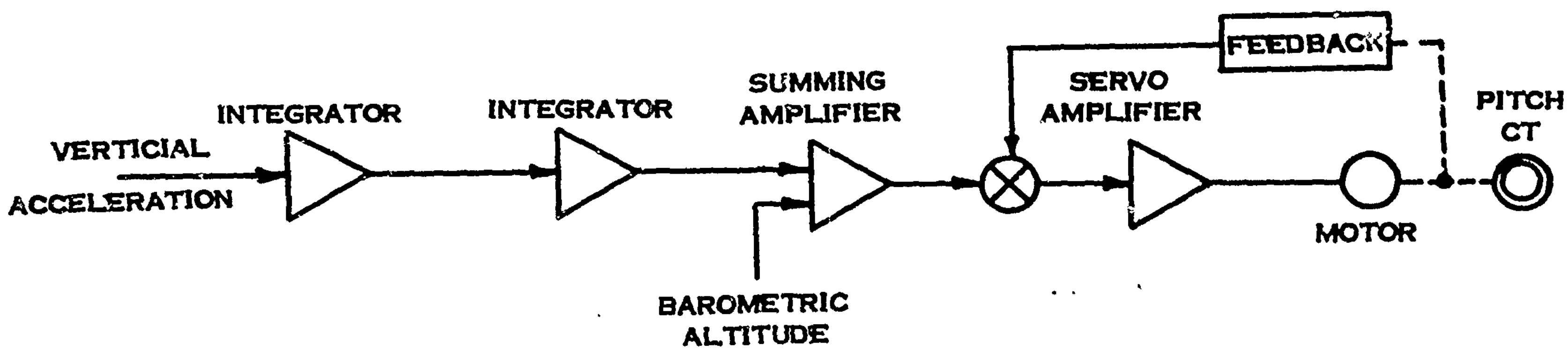
Figure 40 is a simplified functional block diagram of the Teledyne inertial guidance system and the control system for the parawing configuration which has two horizontal stabilizers and one vertical fin. Figure 41 is the block diagram for the lifting body configuration which uses the same guidance system but uses a control system for two elevons.

The two gyros on the platform provide a three axis space reference for the three accelerometers which detect errors in the desired flight path by measuring lateral, longitudinal and vertical acceleration during the vehicle's flight. A computer is necessary to calculate vehicle position relative to earth's coordinates. Attitude reference signals are obtained by torquing the vehicle platform to maintain a local vertical. With such local vertical sensing systems the synchros on the gimbals provide a direct readout of the vehicle attitudes which can be used directly by the flight control system. The attitude signals are used by the servo actuators to maneuver the control surfaces and correct the vehicle's course. Summing amplifiers add the pitch and roll signals so the horizontal stabilizers can control roll and pitch together.

For gust measurements a constant vehicle speed is desirable. Figure 42 is a block diagram of a velocity and range control servo. The vector sums of the platform X and Y accelerometer signals are integrated to obtain vehicle actual velocity. This velocity signal is compared with a preset velocity signal and the difference produces an error signal which controls a servo mechanism operating the engine throttle to increase or decrease vehicle speed. This same block diagram also illustrates one method of shutting off the engine as it approaches the recovery area. The actual distance traveled by the vehicle is obtained by integrating the acceleration twice and comparing the results with a range signal in the computer. When the distance traveled signal becomes greater than the preset range signal, it is amplified to operate a solenoid type propellant shutoff valve.

6.3 COMMAND SYSTEM

There must be some method provided whereby personnel conducting the tests can abort the mission at any point should conditions warrant. This will require a command receiver, decoding circuitry and some type of input into the control systems. By transmitting the proper command from the ground, the launch aircraft or the recovery aircraft, the HI-HICAT propulsion system can be shut



CT: SYNCHRO CONTROL TRANSFORMER

FIGURE 39 ALTITUDE CONTROL SERVO

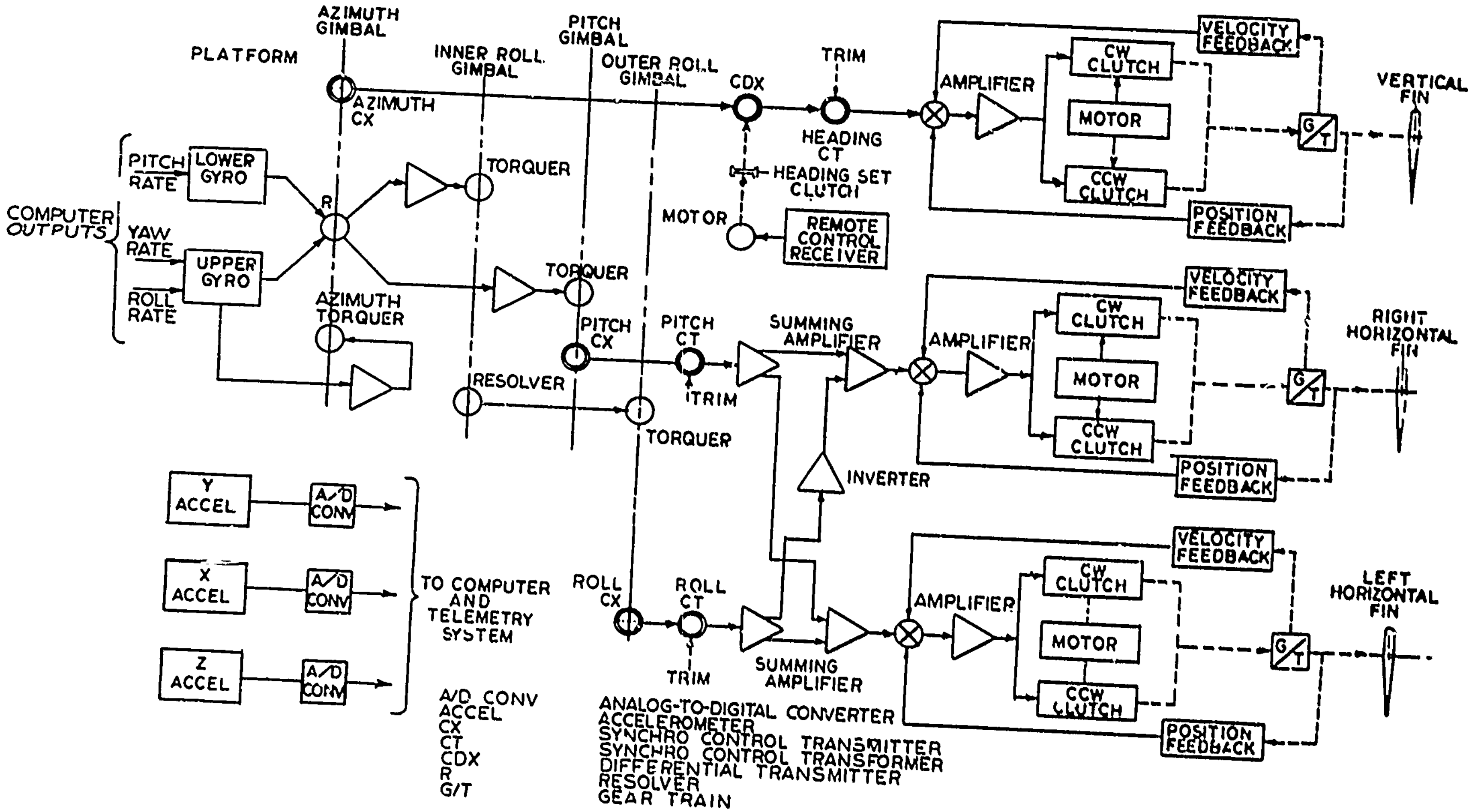


FIGURE 40 PARAWING VEHICLE GUIDANCE AND CONTROL SYSTEM

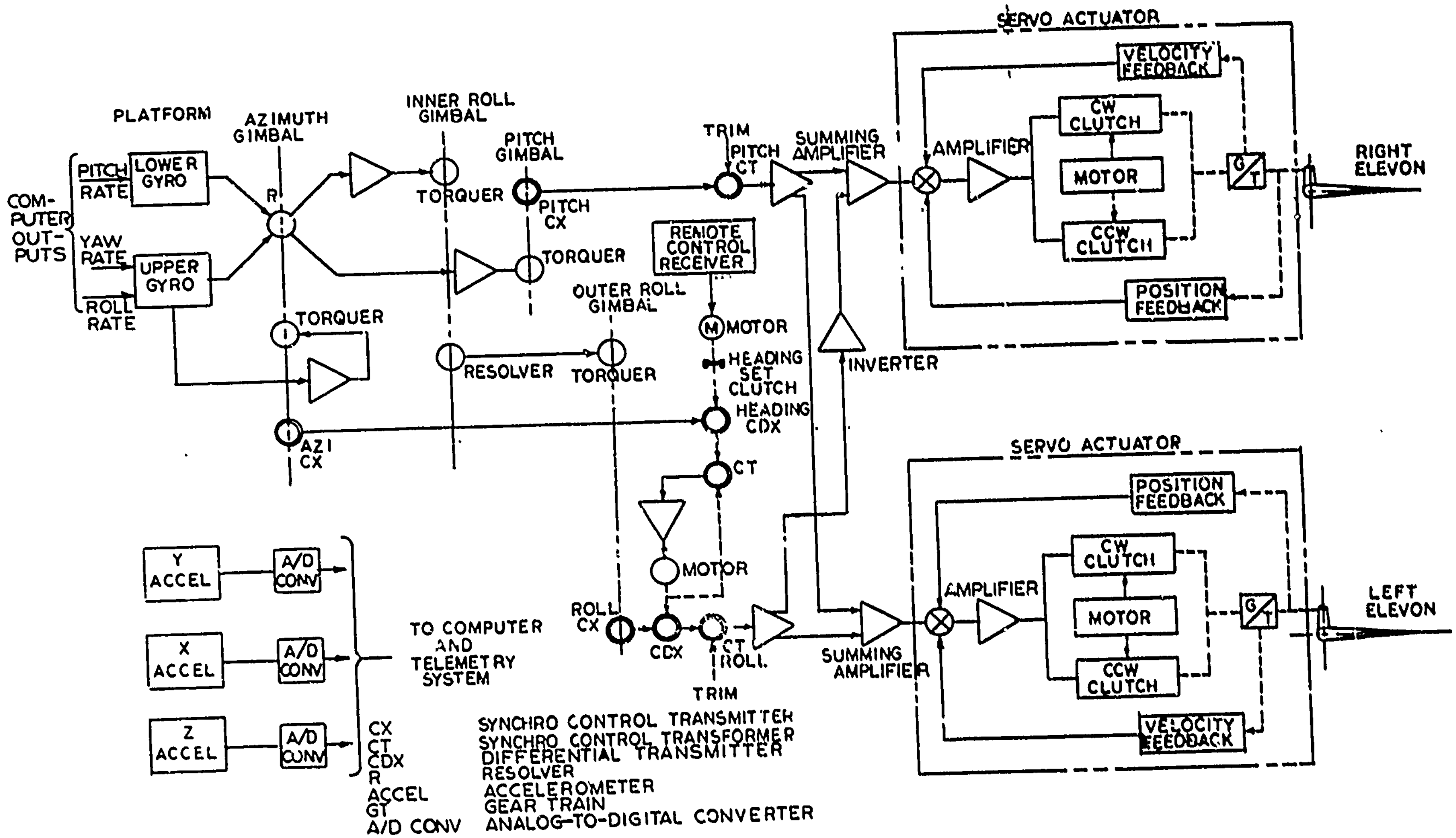


FIGURE 41 LIFTING BODY VEHICLE GUIDANCE AND CONTROL SYSTEM

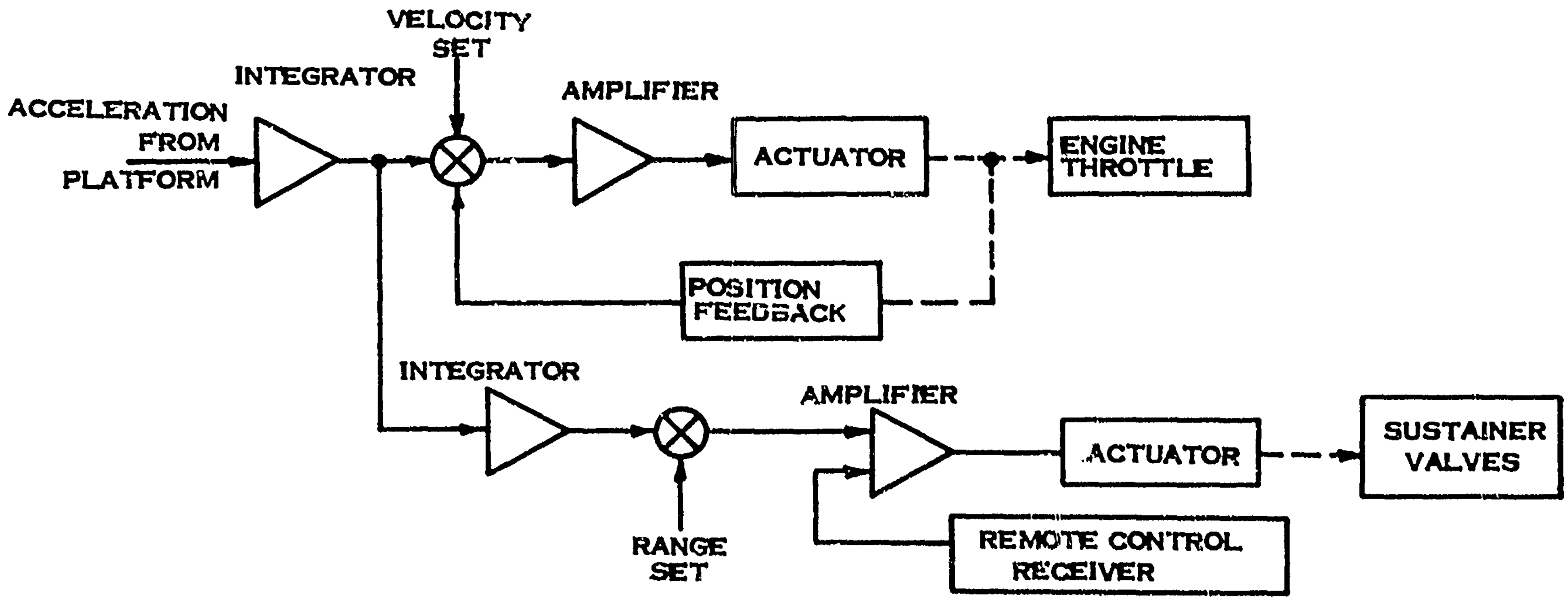


FIGURE 42. VELOCITY AND RANGE CONTROL SERVO

down. Also other commands can be provided as desired but these are presently limited to a heading change and recovery initiation override only. The engine shut down function is required as a safety measure in case the vehicle should fly off course towards a populated area. A block diagram of a command system is given in Figure 43.

Since the command link is required, it would be relatively simple to provide a multiplicity of commands at very little additional cost.

6.4 CONTROL SYSTEM

The control surfaces of the HI-HICAT vehicle are manipulated by individual rotary or linear electro-mechanical servo actuators. The parawing configuration uses three rotary actuators, one for each fin, and the lifting body configuration uses two linear actuators, one for each elevon. The electro-mechanical actuators were selected over other types because of their availability and the ease of supplying power for them. Hydraulic or pneumatic actuators operate on high pressure oil or air supplies which would necessitate the installation of pumps, accumulators, filters, tubing and regulators in the vehicle. All these components are heavy and costly and more difficult to install and maintain than a battery pack. Also hydraulic and pneumatic components are usually custom fabricated for each application.

The Lear-Siegler electromechanical servo actuator is currently available in many sizes and shapes, and the present HI-HICAT vehicle designs employ several of their models. These actuators utilize a magnetic particle clutch. The magnetic clutch is a versatile control element which can be easily controlled by conventional circuits. Clutch type servo actuators have high torque to inertia ratio because the motor and clutch input shaft rotates continuously and only the inertia of the clutch output shaft and load must be overcome during acceleration. Precision control is obtained with a simple controller-amplifier arrangement in which the force output of the actuator is linearly proportional to very low level input signals from the navigating systems.

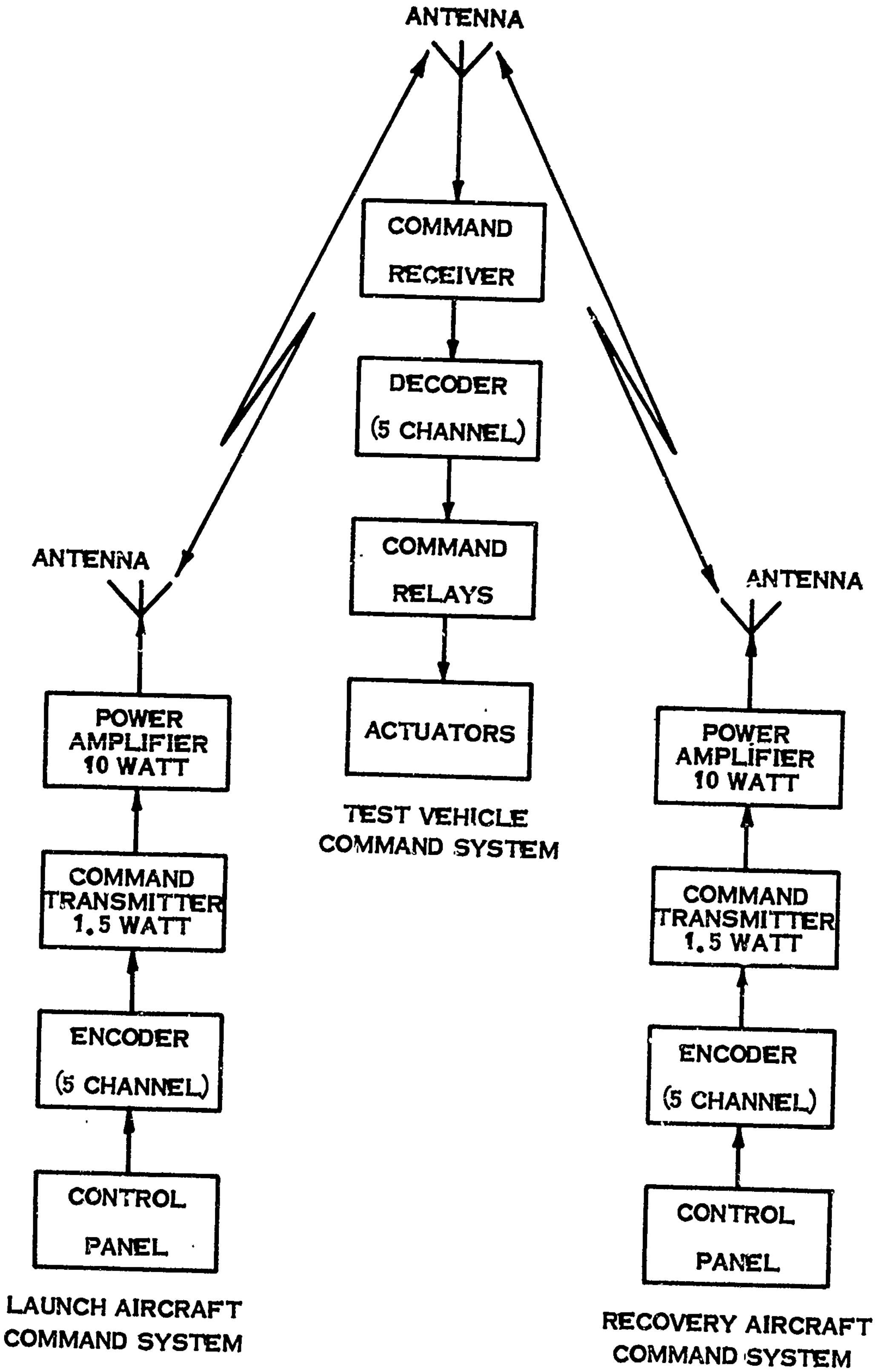


FIGURE 43. COMMAND SYSTEM

SECTION 7

INSTRUMENTATION

7.1 DATA REQUIREMENTS

The purpose of turbulence research is to describe the magnitude and direction of the wind velocities over a certain spectrum of frequencies and to define the meteorological conditions. Turbulence could be measured directly if the direction and magnitude of the relative wind could be measured by fast response instrumentation installed aboard a flight vehicle which is itself not affected by turbulence. Unfortunately, no likely vehicle configuration can be conceived which will not respond to some extent to the long turbulence wavelengths. However, instead of measuring the vehicle motions directly, a system can be imagined with a sophisticated guidance and control system which would control the vehicle in such a manner that it would fly a straight line in spite of turbulence. Such a concept is rejected since the complexity of such a novel, sophisticated guidance and control equipment would greatly exceed the complexity of the extra instrumentation otherwise required.

It is usual to separate the measurement of relative wind into airspeed and flow direction measurements. Customarily, fast response airspeed instrumentation is not installed because of the technical difficulties involved and because past investigations have shown little difference between the longitudinal and lateral turbulence spectrums.

At intermediate and long turbulence wavelengths, the vehicle responds to gusts and such responses must be measured in order to calculate the "true" gust velocity relative to earth coordinates. The motions of a vehicle can be measured by external means such as radar or with internal instrumentation carried aboard the flight vehicle. Radar (and optical systems) must be rejected because atmospheric turbulence and other difficulties prevent detection of the fine scale vehicle motions as discussed in Section 6.1. It will be shown that inertial instrumentation, namely gyros and accelerometers, can provide the desired accuracies.

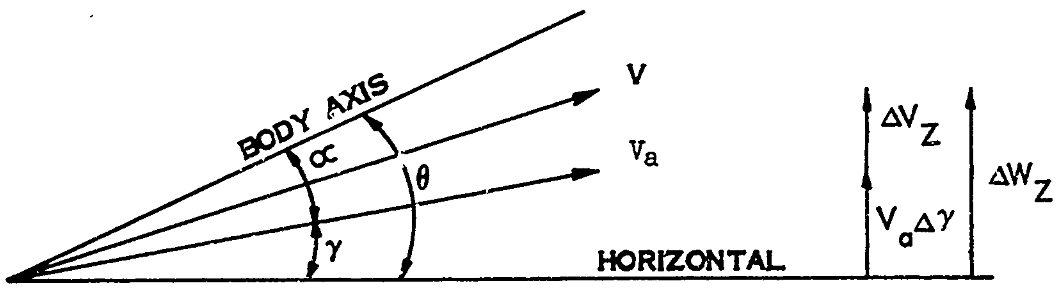
It is also customary to measure both atmospheric pressure and temperature along with the turbulence parameters to better define the meteorological conditions associated with turbulence. Unfortunately, the measurement of atmospheric temperatures has proven impossible to accomplish and no free air temperature instrumentation is included in the HI-HICAT design.

7.1.1 Flow Direction Measurements

7.1.1.1 Accuracy Requirements

The major task to be accomplished by the HI-HICAT system is the measurement of wind gust velocities at altitudes from 70,000 to 200,000 feet. The present contract specifies measuring a 1 fps rms velocity out to a maximum wavelength of 75,000 feet. The minimum frequency of interest can then be calculated from the maximum wavelength by assuming the HI-HICAT vehicle is flying through "frozen" turbulence, a common assumption. The contract does not specify a minimum wavelength or a maximum frequency. In this study a maximum frequency of 10 cps has been chosen in order to obtain a bandwidth comparable to past turbulence investigations.

The error analysis proceeds by assuming the vertical velocities and the attitude angles are small compared to the calm air flight velocities. The design points shall be Mach 4 at 70,000 feet and Mach 6 at 200,000 feet. The diagram below shows the relationship existing between the various velocities and angles involved.



From the diagram, the basic equation describing the kinematic relation between variables can be written:

$$\Delta W_z = V_a \Delta \gamma + \Delta V_z$$

$$\Delta W_z = V_a \Delta \alpha - V_a \Delta \theta + \Delta V_z$$

$$= V_a \Delta \alpha - V_a \Delta \theta + \int_0^t a_z dt$$

where:

a_z = vertical component of body acceleration

t = time

V = vehicle velocity relative to earth axis

V_a = vehicle velocity relative to wind axis

V_z = vertical component of V

W = wind velocity relative to earth axis

W_z = vertical component of W

α = angle of attack

θ = pitch angle

γ = $\theta - \alpha$

and where small perturbations from a calm air flight path are considered.

If the overall accuracy is 1 fps rms, then the accuracy of each term of the equation should be about 0.5 fps rms. The highest design velocity is 6280 fps (Mach 6 at 200,000 feet); hence, the first term of the equation above indicates the flow direction should be measured to an accuracy of

$$\Delta\alpha = 0.5/6280 = 0.00008 \text{ radians} = 0.005 \text{ degree.}$$

The second term leads to a criterion for the accuracy of pitch attitude measurements. Substituting in the second term yields

$$\Delta\theta = \frac{0.5}{6280} = 0.005 \text{ degree}$$

for the precision required. Note that both the flow direction and pitch attitude measurement accuracies are relative (incremental) values. The absolute (bias) error is not of interest.

The third term indicates the accuracy of acceleration measurements. A pure sinusoidal motion is considered with

$$a_z = A_z \sin \omega t$$

where ω is the circular frequency. Substitution in the third term and integration yields

$$\Delta W_z = -A_z \frac{\cos \omega t}{\omega}$$

for the time varying component. Hence the precision required of an accelerometer increases as the frequency decreases. The minimum frequency is

$$\omega = 2\pi f = 2\pi \frac{3880}{75,000} = 0.33 \text{ radians per second}$$

where 75,000 feet is the longest turbulence wavelength of interest and 3880 fps is the minimum design velocity. Therefore,

$$A_z = 0.5 \times 0.33 = 0.165 \text{ fps}^2 = 0.005g \quad (\cos \omega t = 1),$$

an accuracy relatively easy to meet.

Note that the error analysis above is readily adaptable to the horizontal lateral velocities and will lead to the same requirements for accuracy.

The recommended system as discussed in section 7.1.1.3 is capable of measuring .8 fps rms gust @ Mach 4 and 70,000 feet. It also meets the 1.0 fps rms gust requirement at Mach 6 and 200,000 feet.

7.1.1.2 Types of Flow Direction Sensors

Flow direction sensors having a reasonable degree of practicality under high temperature conditions rely on a pressure differential which is measured either directly or indirectly. Rejected immediately is a turbulence sensor being developed which is a variation of the hot wire anemometer where the wires are not exposed but are mounted flush in the surface of a probe. Also rejected

are doppler laser sensors utilizing the frequency shift in radiation back-scattered by atmosphere. Research here is at a very early stage.

The possible types of sensors can be categorized as:

1. pivoted vane
2. fixed vane or Q-ball
3. servoed vane or Q-ball

Arguments are now presented to demonstrate that a servoed Q-ball, similar to that developed and in operation on the X-15 research aircraft, will yield the greatest accuracy.

The pivoted vane can be eliminated from consideration by two arguments: one based on the dynamic response and one based on the minimum possible friction in an "off-the-shelf" readout device. The dynamic response is analyzed by considering a simplified equation for the free motion of a second order system,

$$C_{L_\alpha} \alpha q S \ell + \rho S t_{av} \ell^2 \ddot{\alpha} = 0$$

where:

C_{L_α} = lift curve slope

α = angle of attack

q = dynamic pressure

S = vane area

ℓ = pivot-to-vane length

ρ = density of vane material

t_{av} = average thickness of vane.

The damping term was left out of the above equation because it will have little effect on the order-of-magnitude analysis to follow. Also, note that the mass of the vane support is ignored and that the vane dimensions are considered small compared to the pivot-to-vane length. The above equation then yields a natural frequency.

$$\omega = \sqrt{\frac{C_{L_\alpha} q}{\rho t_{av} \ell}}$$

At frequencies above the natural frequency the response of a second order system starts to drop off rapidly because of inertia.

The lift curve slope at high supersonic speeds can be approximated by the relation

$$C_{L_\alpha} = \frac{4}{\sqrt{M^2 - 1}}$$

which is about one at Mach 4 and two-thirds at Mach 6. For nickel and iron base alloys densities of 0.3 lb per cu in. =16 slugs per cubic foot are typical. A typical vane thickness and pivot-to-vane length might be 3×10^{-3} feet and 1/3 feet, respectively. Using these values the natural frequency becomes $f = \omega/2\pi = 40$ cps at 70,000 feet and 3.3 cps at 200,000 feet.

The response of other "weathercocking" shapes such as a blunted nose cone would probably be worse. Reference 12 presents a frequency response curve for a typical aerodynamically driven sensor with a response worse than that indicated by this analysis.

It might be supposed that the dynamic response could be improved by going to very small vanes but the friction in existing readout devices discourages this approach. The restoring torque of a vane equals

$$C_{L_\alpha} \alpha q S \ell$$

Assuming $\ell = 1/3$ feet as before and $S = 1/50$ sq ft, the torque becomes 0.6×10^{-3} and 0.4×10^{-5} ft-lb at 70,000 and 200,000 feet, respectively, for an angle of attack variation of 0.005 degrees. There are two ways of reading out the pivot shaft attitude angle to the resolution desired over a total angle of attack range from 0 to 50 degrees. These are a two-speed synchro or a special gear arrangement and a precision potentiometer. The torque required for a synchro is less than for a potentiometer. The minimum torque to overcome the bearing friction falls between 2.5×10^{-4} and 2.5×10^{-3} ft-lb. With a gear ratio of 10:1 for the second synchro this becomes 2.5×10^{-3} and 2.5×10^{-2} ft-lb. At 200,000 feet the required values are far removed from the torque available values indicated above.

The above two arguments and the problem of finding suitable vane and pivot materials for the high temperature environment discourages further consideration of moving vanes unless a new and radically different kind of readout device can be invented.

Attention is now directed to fixed and servoed sensors. The technical advantages of a servoed sensor can be demonstrated if the following arguments are considered:

1. For each fixed sensor configuration there is an equivalent servoed sensor.
2. A fixed sensor measures a magnitude while a servoed sensor requires only the detection of a difference in either pressure, force or displacement, either plus or minus. At a wide variety of altitudes and speeds, ample torque is available through the servo device to drive the readout device with a resulting improvement in accuracy.

The above arguments would seem to rule out fixed sensors; however, their greater simplicity must be weighed along with their ability to provide a high degree of resolution at their design dynamic pressure if the maximum quantity to be measured is small. A compromise design has evolved for a one-axis Q-ball which would be servoed for angle of attack measurements only.

Such a design is possible since the angle of sideslip will vary through a much smaller range of values than the angle of attack which has a large nonturbulent component that is dependent upon the cruise attitude. The small range in the angle of sideslip permits employing a differential pressure transducer with a high degree of resolution.

7.1.1.3 The Recommended Flow Direction Sensor

Figure 44 is a drawing of the proposed design. The nose is a 2.5-inch diameter sphere and the afterbody is part of the nose cone of the HI-HICAT vehicle. The sphere is capable of ±25 degree in pitch about a nominal +25 degrees. The array of pressure ports on the sphere is equivalent to that on the X-15 Q-ball. Six pressure ports are used: three differential and one absolute pressure are measured as required in the determination of Mach number, pressure altitude, angle of attack, and angle of sideslip.

The pressure transducers are capacitive types as used on all previous Q-balls. The transducers perform as required for the measurement of flow direction and, in addition, have demonstrated resolution and stability properties consistent with the attainment of the severe 0.005 degree short time accuracy.

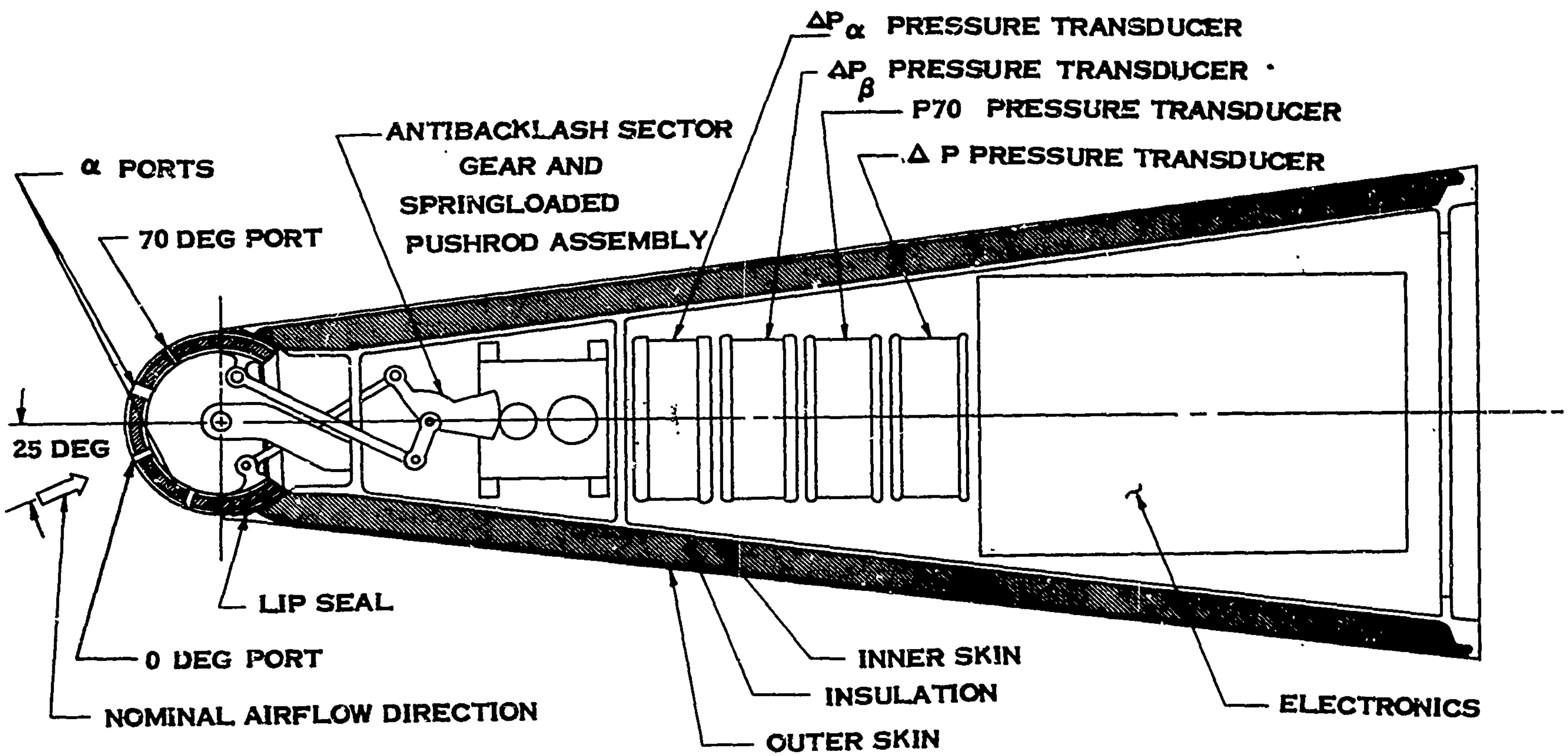
The sphere actuation system is made up of a servo motor driving a gear train whose final sector is pushrod-connected to the sphere. All the gears are spring-loaded to prevent backlash. A synchro is located at the next to the last gear to provide an amplified sphere position readout.

The electronics, power supplies, amplifiers, demodulators and other circuitry are all located in a package aft of the transducers. The components will be mounted on rough boards stacked up within a cylindrical housing. Figure 45 is a block diagram of the electronics. Power required at 28 volts dc (direct current) is 22 watts for the electronics.

Other power requirements will be converted internally from the 28 volts dc. Four hundred cycle power is the primary frequency expected; however, higher frequency excitation may be required for the transducers, depending upon the exact models chosen.

The philosophy to be followed in the electronics is one of utilizing proven components of the most up to date (and additionally, miniaturized) types available. Where integrated circuits are used, they will be of available types: special circuits are not anticipated. Clearly semiconductor devices will be used as active circuit elements throughout. Military quality parts will be utilized.

Servo analysis shows promise of resolving and correlating the angle of attack over a time of 20 seconds within 0.005 degree. The dominant dynamic behavior of the servo system is nominally that of a second order system with a resonant frequency of about 6 cps and a damping ratio of 0.5; thus the amplitude ratio for a sinusoidal input is "flat" within 3 db out to about 7 cps. Hence, the present design does not achieve the desired goal of 10 cps or higher within the limits of already developed components. Study should be conducted to discover if components of new design could substantially improve the response of the Q-ball system.



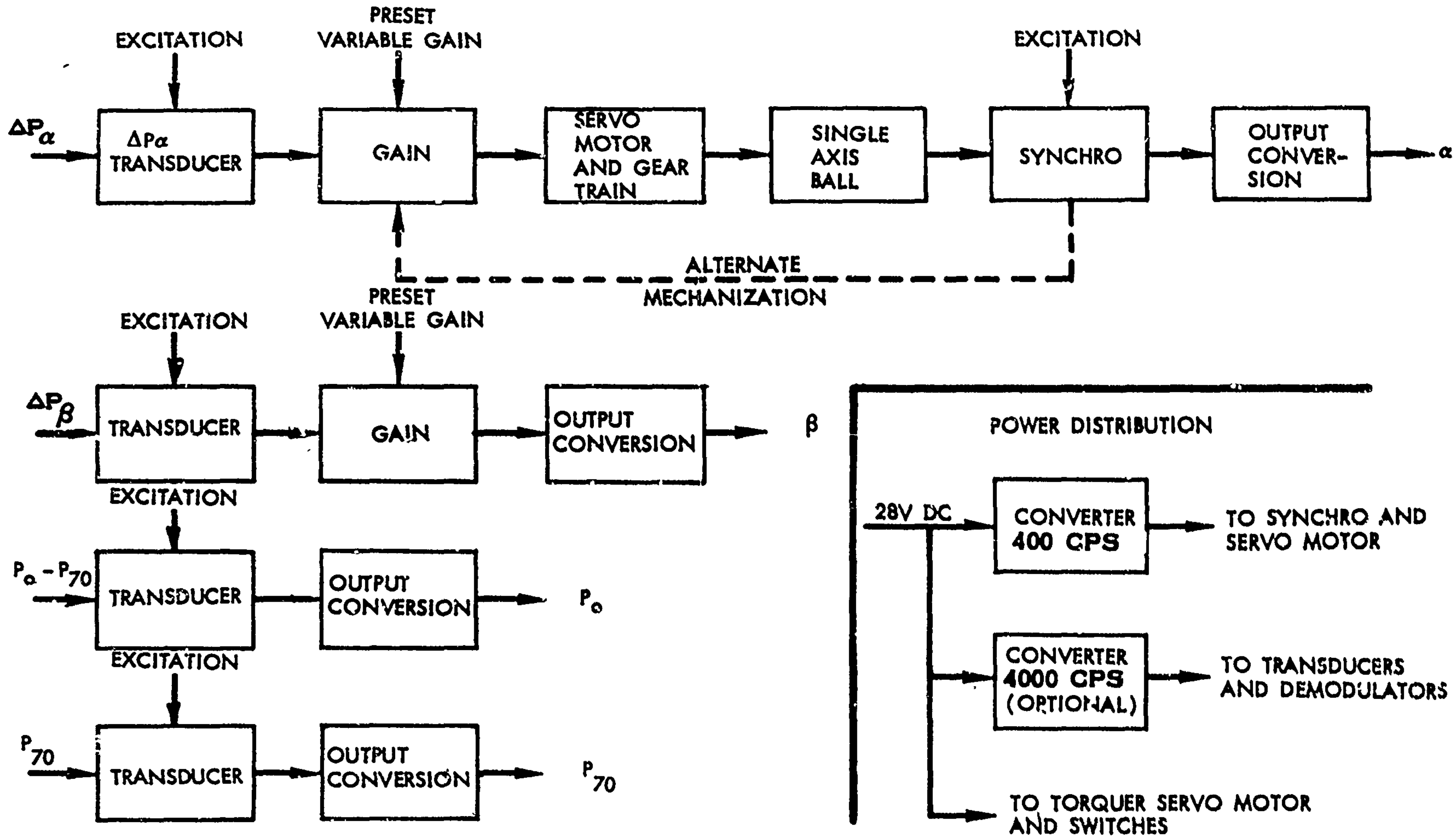
ΔP_α (ΔP_β): DIFFERENTIAL PRESSURE BETWEEN α (β) PORTS

α (β): ANGLE OF ATTACK (SIDESLIP)

ΔP: $P_{70} - P_0$

P_0 (P_{70}): PRESSURE AT 0 (70) DEG PORT

FIGURE 44. Q-BALL DESIGN



(SEE FIGURE 44 FOR DEFINITIONS)

FIGURE 45. SINGLE-AXIS Q-BALL

The Q-ball system is not gain compensated. Since the error signal is a pressure, the anticipated dynamic pressure level must be set prior to flight. The system will operate satisfactorily at dynamic pressures greater or less than the pre-set level by 40 percent.

The sphere rate capability is 320 degrees per second, which corresponds to tracking a wind shear of 5.6 per second. Reference 13 indicates a probability of 99.9 percent that the wind shear is less than 0.06 per second for a wavelength of 985 feet.

7.1.2 Flight Data Measurements

Two pressures on the surface of the Q-ball are measured, an absolute pressure and a differential pressure. The absolute pressure is from a port located 70 degrees from the nominal airflow direction while the differential pressure is between this port and one located at 0 degrees. Since the sphere position is maintained constant relative to the airflow direction, these pressures are independent of the vehicle attitude and their ratio is a unique function of Mach number. Therefore, a solution for Mach number exists. Then, given Mach number and one of the two measured pressures, there are unique closed solutions for other air data parameters, including dynamic pressure and static pressure.

Figure 6 of Reference 12 illustrates accuracy curves derived from X-15 flight test calibrations. The errors are large at the high supersonic cruise speeds anticipated because of the "Mach freeze" phenomenon even when account is taken of the conservativeness of the data. The figure shows errors of 7 and 20 percent at Mach 4 and 6, respectively, for static pressure and errors of 4.5 and 10 percent in Mach number.

The above discussion illustrates the fact that the true speed of the HI-HICAT vehicle should not be derived from Q-ball measurements. Rather, this data must be obtained from the inertial flight data system which is described in detail in Section 6 of this report. The function in the flight data system which generates the flight velocity should be accurate to about 1 percent, assuming the maximum turbulence velocity to be measured is 100 times the minimum.

The other flight data measurements of interest are the pitch, roll and yaw angles and the longitudinal, lateral and vertical accelerations. These flight path parameters could be obtained from the inertial navigation system. However, the accuracy of the synchro transmitters used to read out the attitude angles is an order of magnitude below the instrumentation requirement. Modifications of the existing platform gimbals to add precision transducers would result in a long and costly development program. An alternative is to provide a cluster of precision rate gyros and integrate the signals with ground base computers for the ac attitude angles. The recommended package is a cluster of three high precision instrument sensor gyros and three vernier range extenders used to obtain accurate angular rate data around the pitch, roll and yaw axis of the vehicle. The package contains three Systron-Donner Model 8140 precision servo-torque-balance, dc rate gyros capable of resolving angular velocities of 0.00015 degree per second and three Systron-Donner Model 4106 vernier range extenders. Since $\dot{\theta} = \omega\theta$ and since $\omega = 0.33$ radians per second is the minimum design frequency, $\theta = 0.00015/0.33 = 0.00045$ degree and ample capability is provided for meeting the required accuracy of 0.005 degree.

Quasi-static (dc) attitude angles must also be measured in order to transform the acceleration and wind velocity components from body to earth axis and vice versa. These are obtained, with a satisfactory degree of accuracy, from the inertial navigation system discussed in Section 6. This system is also capable of providing acceleration data to the required degree of accuracy over the entire frequency range of interest, from dc to 10 cps.

7.1.3 Airframe and Subsystem Measurements

To provide a check on airframe and subsystem performance, a number of data channels should be included for engineering parameters such as subsystem pressures and temperatures, structural strains and temperatures and a number of simple on-off type of events. It is expected that all engineering parameters can be instrumented with standard equipment and there will be no requirement for additional development. Extreme accuracies are not necessary. The engineering parameters to be measured can be reduced as the program advances through the development flight tests until a full set of meteorological parameters can be measured. In this manner the number of channels required can be reduced to a minimum.

7.2 DATA CONDITIONING EQUIPMENT

By proper selection of transducers, excitation voltages, and pre-conditioning, the transducer outputs can all be developed as a dc voltage which is a linear function of the measured parameter. The dynamic range and scale factor of each of these signals must now be adjusted to fall within the 0 to 5 volt input range requirement of the multiplexing equipment. Isolation is also needed to prevent electrical loading of the transducers which would adversely affect their accuracy. There are three generally acceptable methods by which this can be accomplished. One approach would be to reduce the amplitude of all signals, by using resistor divider networks, to a 0 to 20 millivolt range and use low level commutation followed by a single amplifier to establish all commutated (multiplexed) data at a 0 to 5 volt range. This method is not considered to be optimum since low level signals cannot be measured to a high degree of accuracy due to inherent system noise. A second method would be to supply a dc isolation and scaling amplifier for each measured parameter to properly scale the measured parameters to be within the desired 0 to 5 volt scale for high level commutation. This method would provide for optimum system accuracy and performance; however, it is also the most expensive due to the extra equipment and power required. The third method is a compromise which would provide for both high and low level commutation within the telemetry module and condition the signals which do not need to be 20 millivolts or 0 to 5 volts. In this case low level signals which do not need to be read to extreme accuracies can be low level commutated with a resulting reduction in equipment. This method is recommended as being the optimum for the HI-HICAT system.

A typical bridge circuit followed by an isolation and scaling dc amplifier is illustrated in Figure 46. In some cases the overall dynamic range of measurement of a parameter is too great for sufficient resolution to be obtained from a single output range of only 0 to 5 volts. In these cases it is recommended that a range extender be employed to provide both a coarse and a fine data channel. Figure 47 illustrates the major components of the range extender unit. The range extender functions as follows. The output of the transducer

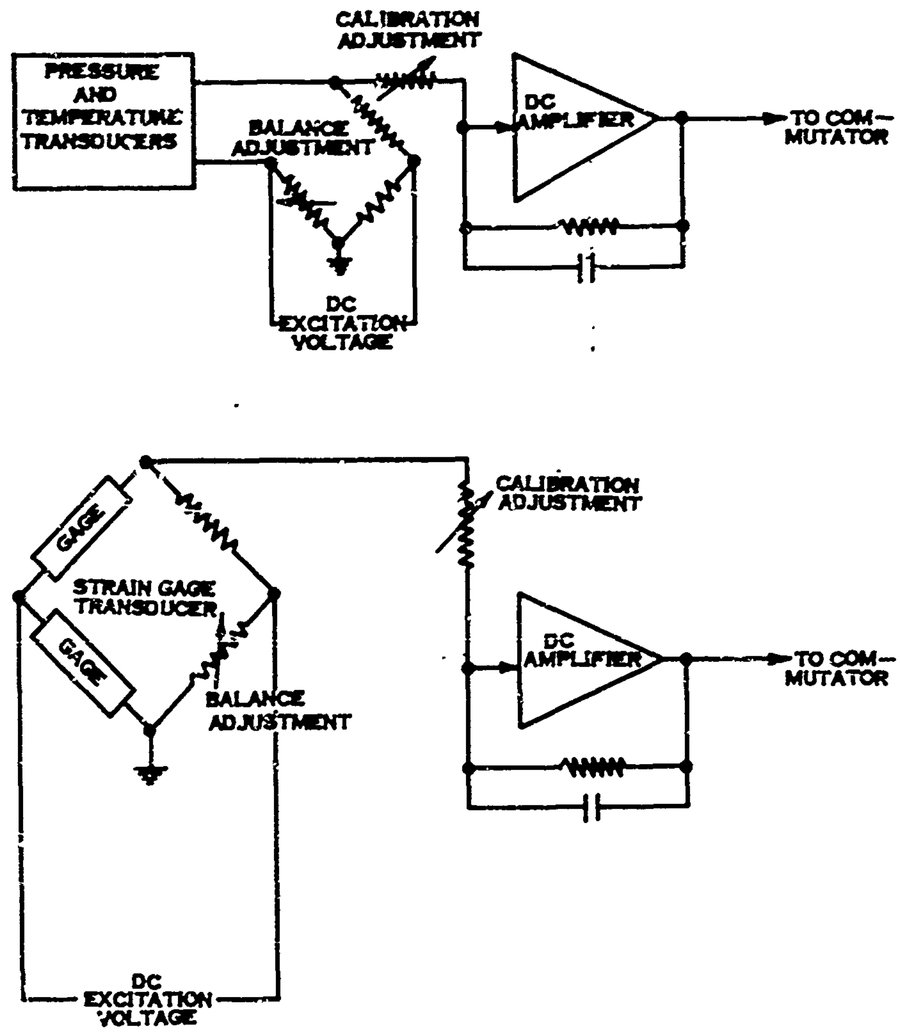


FIGURE 46 TRANSDUCER OUTPUT SIGNAL CONDITIONING

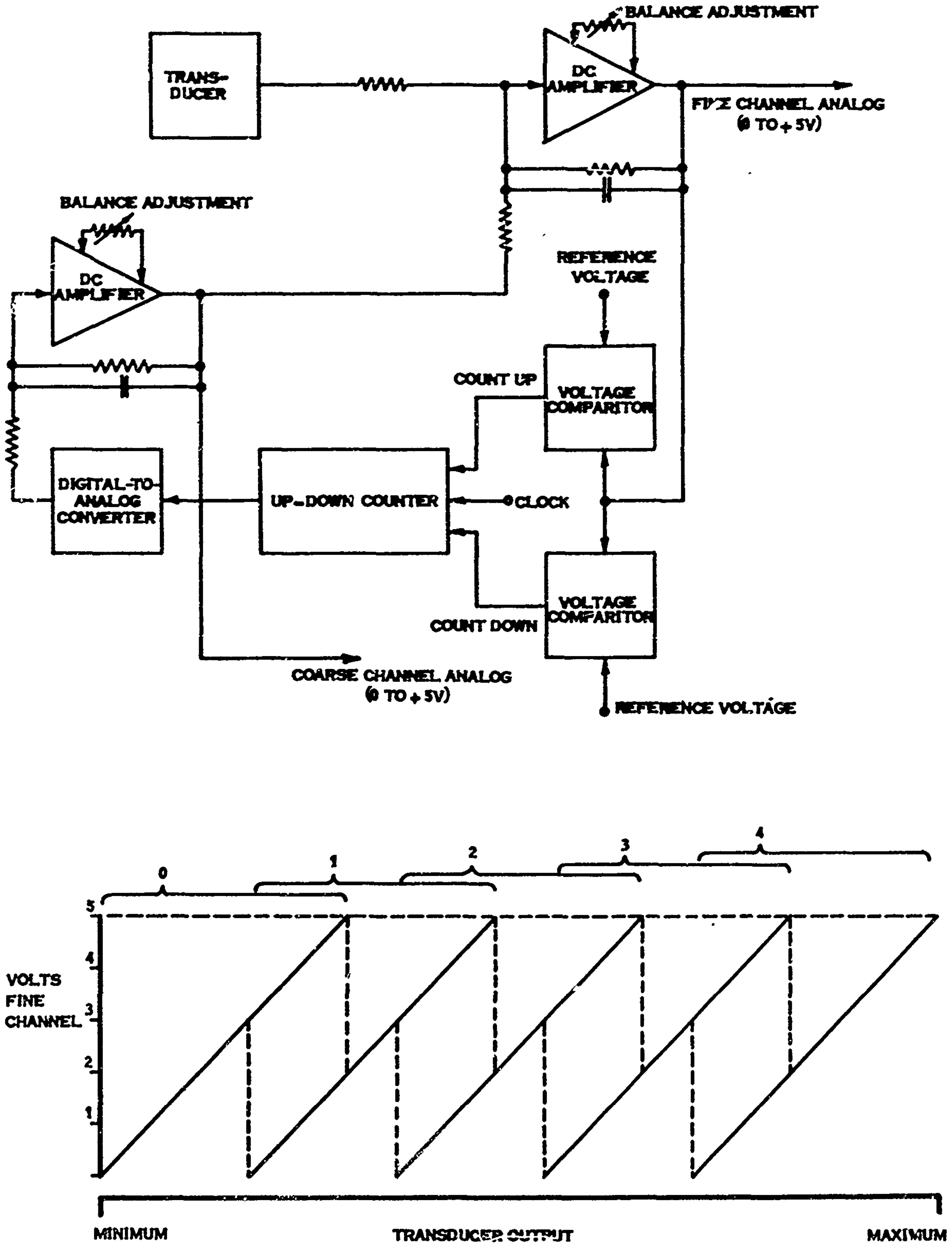


FIGURE 47 HIGH RESOLUTION RANGE EXTENDER

is amplified by a dc amplifier. If no additional circuitry were used and the transducer output varied over a wide range, the output of the amplifier would be quickly driven from saturation to cut off. However, as the output of the amplifier approaches either limit, one of the voltage comparator circuits will function to open a count up or a count down gate in the up-down counter. This action will allow a clock pulse to drive the counter either up or down depending on which amplifier limit is being approached. The output of the counter is fed to a digital-to-analog converter whose output is fed through a dc amplifier and a summing resistor to the input of the first dc amplifier. The polarity of the converter output is selected to be opposite to that of the transducer output. The action of this circuitry is to shift a small, amplified range of measurement anywhere within the large dynamic range of the measured variable. Thus by observing the converter output as a coarse data channel and the operational amplifier output as a fine data channel, accurate measurements over large dynamic ranges can be made with a minimum data channel resolution as illustrated in Figure 47. The only limiting factor to the overall resolution is the system noise level.

7.3 DATA STORAGE AND TRANSMISSION

7.3.1 General Requirements

Once having provided the instrumentation necessary for the measurement of all pertinent parameters to the accuracies stipulated by the mission requirements, it becomes necessary to provide some means of transferring these data to the ground for complete data reduction and analysis. These data can either be transmitted directly to the ground in real time, or data can be stored in some fashion aboard the vehicle and recovered on the ground at the conclusion of the mission. Regardless of the methods used, care must be exercised to assure that the transmittal method selected does not in any way deteriorate the accuracy of these data.

Since the vehicle is to be recovered at the end of each mission, it would be possible to store the data aboard the vehicle. The obvious objection is the possible loss of all data should the vehicle be lost. A telemetry link with the ground could be used which would assure the reception of at least some data whether the vehicle was recovered or not. However, this method requires more equipment either on the ground or in the launch and recovery aircrafts, or both. Both methods are considered in the following analysis.

7.3.2 On-Board Data Storage

Digital data storage for a 10 to 15 minute flight requires a capacity of better than 13 million bits. Although there are numerous methods and equipment available for the storage of digitized data, only the magnetic tape recorder is considered practical because of the large volume of data. Magnetic tape recorder technology is well advanced and there are a number of units available which have been especially designed to withstand environmental conditions far more severe than those expected for the HI-HICAT mission.

If the data were to be recorded in analog form, a typical tape recorder unit, supplied with record only electronics, would occupy a volume of about 0.6 cu. ft., weigh about 20 pounds, and use from 30 to 35 watts of power. This would be a machine supplied with a 5 inch reel of one inch tape and providing 14 data tracks. Synchro signals would be directly recorded while temperature and pressure signals would be time shared on one track. A five inch reel of tape provides about 26 minutes of recording time at a tape speed of 7-1/2 ips and would be a more than adequate margin of data storage. A tape speed of 7-1/2 ips allows for a direct recording bandwidth of 100 cps to 25 kc and with a FM carrier of 13.5 kc, would provide a bandwidth of from dc to 2.5 kc. Such a bandwidth is more than adequate for the HI-HICAT mission.

All considerations to this point favor analog recording of HI-HICAT data. There is, however, one very important limitation which rules out this method of data storage. The average signal-to-noise ratio is from 32 to 40 db. For the most optimum signal-to-noise ratio, this would limit the readable dynamic range for any recorded parameter to 100 to 1. This range is less than that required to obtain the accuracy stipulated for wind gust measurements by over a factor of 10. Hence analog recording is not recommended.

Another approach is to record the data in digital form, thereby overcoming the objectionable noise level of the analog recording method. With proper selection of bit rate, word length and sampling rate, the recorder just described could be used. However, with all data reduced to a digital pulse train, a much smaller recorder will serve. A two track machine using a double pass on 450 feet of 1/2 mil mylar tape at a tape speed of 10 ips would provide a recording time of 18 minutes. One track would be used for data and the other for a clock pulse at a packing density of 1000 bits per inch of tape. Such a recorder would have a maximum size of 7.5 by 4.5 by 3.0 inches, would weigh less than 5 pounds and require only 5 watts of power. The missed bit error rate would be less than 1 in 10^5. Such a recorder would be about the size and weight of a telemetry transmitter, but it would require much less power. The tapes could be played back and the data reduced at any number of existing Air Force data reduction centers thus eliminating any need for additional ground support equipment.

There would seem to be no valid reason against airborne data recording in digital form once the HI-HICAT system has proven itself reliable. For the early development flights, though, a telemetering transmitter should be installed and flights made over a missile range where ample ground support equipment and facilities are available. Later flights with an airborne recorder could be made in any nonrestricted area in the world.

7.3.3 Telemetry Link

7.3.3.1 Range Limitations

One of the first considerations is range. A telemetry signal generated in the HI-HICAT vehicle is of little use if it is not received. There are three places where receiver sites may be located: the launch aircraft, the recovery aircraft, and on the ground. Maximum range for any ground station is approximated to reasonable accuracy by

$$d = (2h)^{1/2}$$

where

h = transmitter height above sea level in feet and

d = radio distance to effective horizon in miles.

From this equation, the maximum reception range, when the HI-HICAT vehicle is flying at 70,000 feet, is about 370 miles. At 200,000 feet, it is about 640 miles.

As the figures show, if ground stations are used for reception of telemetry data, flight will be restricted to areas in the immediate vicinity of existing receiver sites. This could be overcome by providing one or more transportable receiver systems, but at considerable expense. If the flights are over water, the problem becomes even more complex, for receiver sites must now be provided on ships stationed in the test area. A much more flexible method is to provide a receiver and a tape recorder in both the launch and recovery aircraft, using any available ground receiver site that is in range as backup.

7.3.3.2 Transmitter Power

For normal free-space propagation, an equation to compute the transmitter power for a required output signal-to-noise ratio is given below in terms of antenna reflector dimensions and system parameters.

$$P_t = \frac{\beta_1 \beta_2}{40} \frac{BL^2}{f^2 r^4} \frac{F}{K} \frac{S}{N}$$

where

P_t = power available at the transmitter output in watts,

β_1 = power loss ratio due to transmission line at transmitter,

β_2 = power loss ratio due to transmission line at receiver,

B = root-mean-square bandwidth in megacycles,

L = total length of transmission in miles,

f = carrier frequency in megacycles,

r = radius of parabolic reflector in feet,

F = power ratio noise figure of receiver,

K = improvement in signal-to-noise ratio due to modulation and,

$\frac{S}{N}$ = required signal-to-noise ratio at the receiver.

It is difficult to determine exact values for the power loss ratio without a complete system design, but a value of 6 db (0.5) can be selected as a worst case value. A typical system value for B is 0.1 mc, for L is 600 miles maximum, for f is 250 mc, for r is 0.5 feet (this value for r was selected

since a parabolic reflector is roughly equivalent to a half wave dipole), for F is 5, for K is 0.2 (FM system) and for S/N is 4 (for consistent data). Substituting in the above equation indicates a minimum usable transmitter output is 1/2 watt. Transmitters are available in almost any power range up to 20 watts. To provide a reliable signal to the receiver and a reasonable design factor, at least a 5 watt transmitter is recommended.

7.3.3.3 Antennas

When calculating the minimum required transmitter power above, both the transmitter and receiver antennas were taken as half wave dipoles. A slot or flush cavity antenna could be used without any projecting elements to disturb the aerodynamics of the HI-HICAT vehicle. A properly designed slot or flush cavity antenna compares very favorably with a dipole both in radiated pattern and efficiency.

The launch and recovery aircrafts should be equipped with slot, flush cavity or half wave dipole antennas. Additional reliability could be achieved with a directional antenna, but the additional complexity does not seem to be warranted.

7.3.4 Data Coding

Since there is such a large number of parameters which must be measured, it is not practical to provide a continuous channel for each. Some method of time sharing is necessary.

If each of the high accuracy data parameters is measured at a rate of at least 40 samples per second, this will provide a high frequency response satisfactory for turbulence research. This rate provides a data sample once every 150 feet of vehicle travel at Mach 6.

There are three major ways that sampled data can be coded. The values may be coded as a pulse amplitude (PAM), as a pulse duration (PDM) or as digital information (PCM). The first two ways are limited to a dynamic range of about 100 to 1 which is very difficult to improve upon because of the effects of noise and bandwidth limitations of normal telemetry systems. PCM does not have this serious limitation since the data reduction equipment is only required to determine which of two conditions exist for each pulse period. These conditions can be the presence or absence of a pulse, a wide or narrow pulse, a positive or negative pulse, or one of two discrete pulse amplitudes. The most common coding method in use today is the non-return-to-zero absence or presence of a voltage level for the entire pulse period. This method, although somewhat more difficult to decode because of synchronization difficulties, utilizes the available time to the greatest degree and is recommended for the HI-HICAT system.

Once the data has been converted to its digital equivalent, the accuracy will not deteriorate, provided the data is properly received and decoded. Accuracy depends only on the ability to convert to a digital code and on the selected word length. A data word length of 10 bits will provide a dynamic range of over 1000 to 1. This is considered to be optimum for the HI-HICAT system. The accuracy of transducers, the normal noise levels expected and the difficulties experienced in analog-to-digital converter design restricts going to a greater word length. The actual word length will be 13 bits since one extra bit acts as a parity check and two bits are for word synchronization.

Having now established a rate of 40 samples per second and a word length of 13 bits, a bit rate can be determined. The total words per frame, which corresponds to the number of data channels, is estimated to be 20. The bit rate equals the sample rate times the word length times the number of words and leads to a rate of 10,400 bits per second.

Many of the parameters need not be measured at a rate as high as 40 samples per second. Most temperature measurements and other system parameters can be sampled at a much slower rate and subcommutated onto a single channel. Ten such parameters could be measured at about 4 samples per second, thus occupying one channel (word) in the main frame. Three such subcommutated words should be sufficient for the HI-HICAT mission.

7.3.5 Airborne Equipment

The major components of the data acquisition and transmission system are shown in Figures 48 and 49. The transducers, signal conditioning equipment, transmitter, and antenna have been discussed. The only remaining components fall within what is termed the telemetry module, as shown in Figure 49. Although this package, containing the commutators or multiplexers, sample-and-hold circuits, analog-to-digital converter and programmer, is not available as an off-the-shelf item, very similar equipment has been built in modular form for use on various missile programs. Only the proper selection of a module and minor packaging modifications are required to supply a subsystem which will meet the requirement of the HI-HICAT program.

The functions of the telemetry module are to select, in a programmed fashion, each parameter to be measured, sample the output and generate the binary number equivalent to the sample value. This digital information is then read out in serial form as a straight binary, non-return-to-zero pulse train for transmission or recording or both. The complete sequence of events necessary to accomplish the above is controlled by the programmer, which contains a precision clock oscillator for accurate timing.

After a data sample has been digitized, the digital information is stored temporarily in an output register which forms a part of the analog-to-digital converter. The programmer then generates the parity bit and the two bit word synchronization signal and adds this information to the register to form the composite word. The word is then read out to the transmitter or recorder in serial form. As these data are read out, the next parameter is sampled and digitized. The sequence of events is continuous such that the output is a continuous pulse train of data in the correct format.

7.4 DATA REDUCTION

Regardless of whether the data is recorded onboard the HI-HICAT vehicle or is telemetered to the ground via a VHF radio link, the ground readout equipment is similar. Figure 50 shows the major items required for ground readout.

The data is recorded or transmitted in straight binary. This may or may not be compatible with the input requirements of the computer which is used for data reduction. However, after the data pulse train has been normalized (filtered, detected and reconstructed) it can be converted to a binary coded decimal form or to any other more suitable code which is compatible with the input requirements of the computer system to be used for data reduction and evaluation.

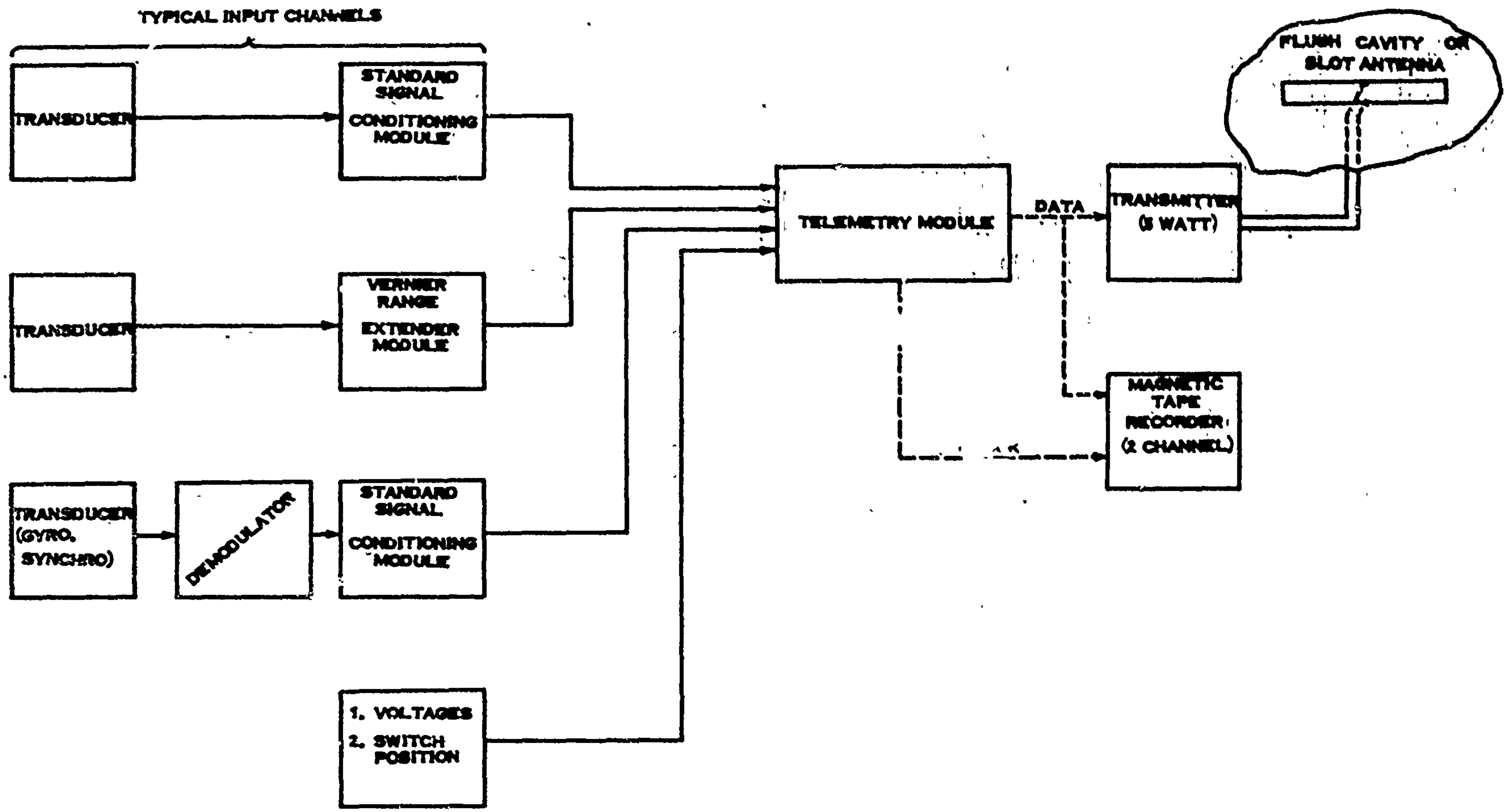


FIGURE 48 AIRBORNE INSTRUMENTATION

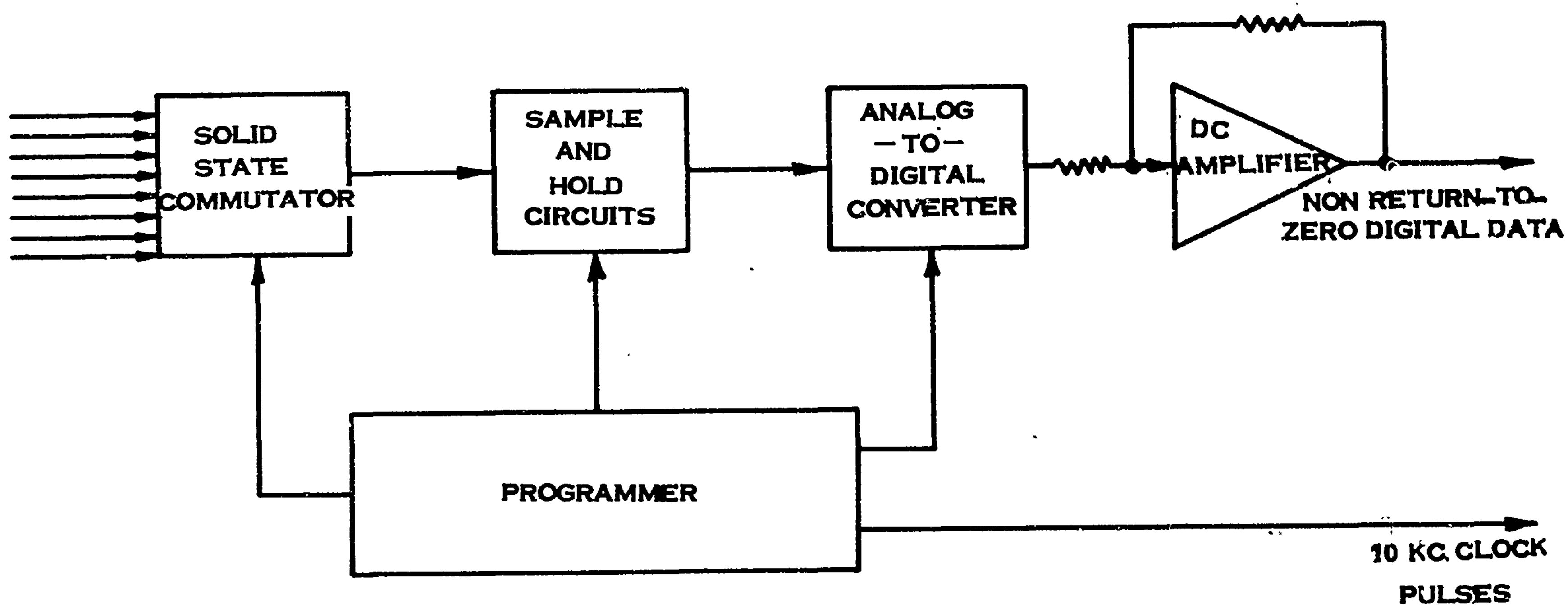


FIGURE 49 TELEMETRY MODULE

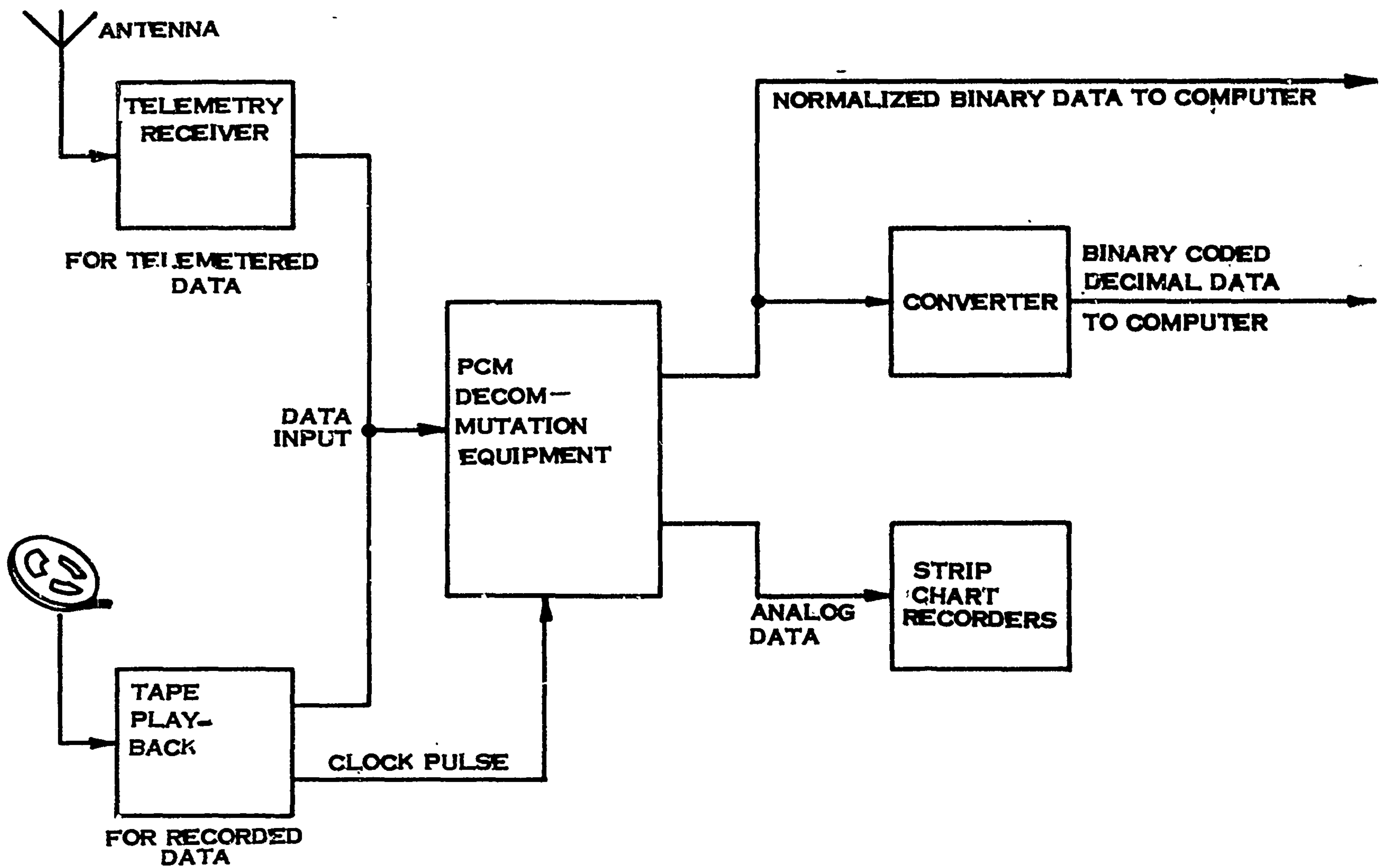


FIGURE 50 GROUND READOUT EQUIPMENT

Since the equipment required for the ground readout and analysis of data are available at a number of Air Force data reduction centers, additional equipment is not considered necessary to support the HI-HICAT program. Should a desire arise for a special data reduction center exclusively for the HI-HICAT program, off-the-shelf equipment is available from a number of suppliers.

7.5 POWER REQUIREMENTS

The source of power for the HI-HICAT vehicle must be compact, yet capable of great capacity; lightweight, yet capable of high current drains. The most ideal source is a battery pack made up of silver-zinc rechargeable cells. Such units are available with energy outputs of 40 to 50 watt-hour per pound and 2.4 to 3.2 watt-hour per cubic inch. The average cell voltage under load is about 1.4 volts; therefore, a battery made up of 20 cells will have an average voltage under load of about 28.0 volts. From full charge to the discharged state the voltage should not change by more than 4 volts. 115 watts of battery power will supply a satisfactory reserve for instrumentation requirements.

7.6 INSTRUMENTATION LIST

The instrumentation package will include a time shared digital data system providing 20 channels (words) in the main frame. One of these main frame channels will be the frame synchronization word and three channels will be subcommutated leaving 16 main frame channels as data channels for data which is to be sampled at a high rate. Each of the three subframes will include 11 subchannels. One each of these will be used as a subframe synchronization word leaving 10 channels for quasistatic data channels which are sampled at a relatively slow rate. Two of these subchannels, in turn, shall provide 20 binary channels for on-off functions. An instrumentation list showing range, sampling rate, maximum resolution, and type of transducers is given in Table 4, which would be applicable to turbulence data gathering flights. Channels 17, 18 and 19 are given over entirely to engineering parameters and the ones listed are a typical set.

TABLE 4

INSTRUMENTATION LIST

TM CH	SUB CH	PARAMETER	RANGE	SAMPLE RATE	MAXIMUM RESOLUTION	TRANSDUCER TYPE
1		Pressure Altitude	0 to 2000 PSF	45/Sec	2 PSF	Pressure
2		Dynamic Pressure	0 to 1000 PSF		1 PSF	Q-Ball Sensor
3		70 Deg. Port Pressure	0 to 100 PSF		0.1 PSF	
4		Angle of Attack Coarse	0 to 50 Deg		0.05 Deg	
5		Angle of Attack Fine	0.05 Deg		0.005 Deg	
6		Angle of Sideslip	±2.5 Deg		0.005 Deg	Q-Ball Sensor
7		Pitch Rate	2 Deg/Sec		0.002 Deg/Sec	Rate Gyro
8		Yaw Rate	2 Deg/Sec		0.002 Deg/Sec	Rate Gyro
9		Roll Rate	2 Deg/Sec		0.002 Deg/Sec	Rate Gyro
10		Pitch Angle	0 to 50 Deg		0.05 Deg	Position Gyro
11		Yaw Angle	±25 Deg		0.05 Deg	Position Gyro
12		Roll Angle	±25 Deg		0.05 Deg	Position Gyro
13		Vertical Acceleration	0 to +2.0G		0.002G	Accelerometer
14		Lateral Acceleration	±1.0G		0.002G	Accelerometer
15		Longitudinal Acceleration	±1.0G		0.002G	Accelerometer
16		Spare	---	45/Sec	---	---
17		Battery Voltage	0 to 32V	4/Sec	0.03V	None
	1	Position	N/A*	4/Sec	N/A*	Synchro

*Not applicable to meterological data

TABLE 4 (CONTINUED)

TM CH	SUB CH	PARAMETER	RANGE	SAMPLE RATE	MAXIMUM RESOLUTION	TRANSDUCER TYPE
	2	Position	N/A*	4/Sec	N/A*	Synchro
	3	Position	N/A*		N/A*	Synchro
	4	Position	N/A*		N/A*	Synchro
	5	Magnetic Heading	360 Deg		0.4 Deg	Magnetometer
	6	400 CPS Converter Frequency	400 ±20 CPS		0.4 CPS	Frequency Detector
	7	400 CPS Converter Frequency	115 ±10V		0.1V	Rectifier
	8	Reference Voltage	20 ±2V		0.02V	None
	9	Battery Voltage	0 to 32V		0.03V	None
	10	Spare	---		---	---
	11	Subframe Synchronism	---		---	---
18	1	Temperature	N/A*		N/A*	Thermistor or Thermocouple
	2					
	3					
	4	Temperature				Thermistor or Thermocouple
	5	Pressure				Pressure
	6					
	7					
	8	Pressure				Pressure
	9	10 Binary Channels				Switches
	10	10 Binary Channels	N/A*		N/A*	Switches
	11	Subframe Synchronism	-	4/Sec	-	---

*Not applicable to meterological data

TABLE 4 (CONCLUDED)

TM CH	SUB CH	PARAMETER	RANGE	SAMPLE RATE	MAXIMUM RESOLUTION	TRANSDUCER TYPE
19	1	Temperature	N/A*	4/Sec	N/A*	Thermistor or Thermocouple
	2					
	3					
	4	Temperature				Thermistor or Thermocouple
	5	Strain				Strain Gage
	6					
	7					
	8	Strain	N/A*		N/A*	Strain Gage
	9	Spare	-		-	---
	10	Spare	-		-	---
	11	Subframe Synchronism	-	4/Sec	-	---
20		Frame Synchronism	-	45/Sec	-	---

*Not applicable to meterological data

SECTION 8

PRESSURIZATION AND COOLING SYSTEM

A schematic of a proposed pressurization and cooling system is presented in Figure 51. Helium storage at extremely low temperatures and supercritical pressures yields fluid densities upwards of twice that for liquid helium which is 7.8 pounds per cubic foot. Compared to liquid nitrogen and other pressurizing gases, it offers a lighter system without the problems that can exist with a two-phase fluid. Such systems have been developed for various projects during the last few years and their cost is not considered excessive for the HI-HICAT vehicle.

The analysis to follow was completed for a parawing vehicle but is applicable with only slight modification to a lifting body vehicle.

8.1 INSTRUMENTATION COMPARTMENT COOLING

A jet pump will be used to circulate the helium in the instrumentation compartment. One quarter-inch of silica fiber insulation is required to reduce the aerodynamic heat load from 130,000 Btu per hour to 13,000 Btu per hour. The outlet temperature of the instrumentation package is maintained at a maximum 175°F. Primary helium gas enters the jet pump initially at -430°F, but by the end of cruise the helium will be at -70°F because the storage tank pressure must be maintained at 1000 psia by internal heaters. For this analysis the temperature of the helium injected into the pump was assumed to be -250°F. For a jet pump secondary-to-primary flow ratio of 2.0, a helium injection rate into the pump of 28.5 pounds per hour is required. Secondary recirculating helium from the compartment is drawn into the pump at a rate of 57 pounds per hour and 300°F. The helium exits from the mixing section at 85.5 pounds per hour and 115°F.

8.2 ACTUATOR COMPARTMENT COOLING

Cooling of the actuator compartment and equipment is similar to that discussed above for the instrumentation compartment. The cooling requirement is slightly larger because of the large equipment heat load. A jet pump flow ratio of 2.0 was assumed, and the primary helium was assumed to be at -250°F. For these conditions, helium is injected into the pump at a required 35 pounds per hour. Secondary, recirculated helium is drawn into the pump at 280°F and 70 pounds per hour. The helium leaves the pump at 104°F and 105 pounds per hour. For the assumed equipment heat load, the equipment helium exit temperature will be 250°F.

8.3 THERMAL PROTECTION FOR THE PROPELLANTS

The maximum bulk temperature for IRFNA is 140°F whereas for Hydyne MAF-4 the limit is 335°F. A significant amount of heat shield material will be required, but due to the complexities involved, no detailed analysis has been attempted at this time. The present design includes an average thickness of 1/4 inch of cork for the parawing tanks on the basis of preliminary estimates and 1/4 inch of silica fiber insulation for the lifting body tanks.

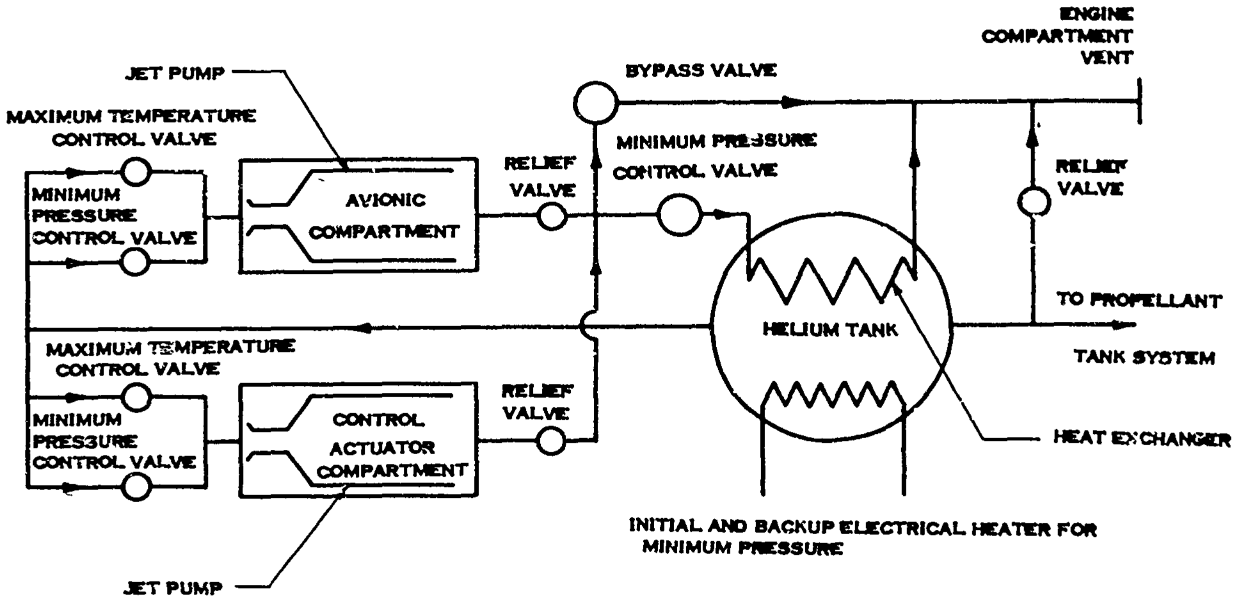


FIGURE 51 PRESSURIZATION AND COOLING SYSTEM

8.4 PROPELLANT TANK PRESSURIZATION

One major difference between the lifting body and the parawing vehicle is the use of a pump in the former to boost the pressure of the propellants up to a level of 620 psia. To achieve the required propellant loading, the propellant tanks for the lifting body must occupy all usable space available. Complex shapes result, which are inappropriate for a pressure-fed system. Therefore, the tanks are pressurized to only 35 psia to provide positive inlet pressure to a turbopump as compared to a maximum pressure of 465 psia for the parawing tanks. The pump is located under the recovery system compartment, and it delivers propellants to the engine module at approximately 620 psia. The propellant tanks in either case will be pressurized by gaseous helium. Storage tank pressure is initiated with an electric heater, but once heat is available from the helium exhausting from the cooled compartments, the tank pressure is maintained by the heated helium passing through a heat exchanger inside the tank.

SECTION 9

WEIGHT

The weight and balance for both the parawing and the lifting body vehicle are given in Tables 5 and 6. The parawing vehicle is 25 feet, 4 inches long, about the maximum feasible for installation under an F-4C aircraft. The 19-inch diameter, however, is somewhat arbitrary. By increasing the diameter, the mass ratio can be increased with a resulting increase in performance, but the increase is small compared to the increase in overall weight. An increase in overall weight means an increase in acquisition and operating costs. The present diameter is believed to strike a satisfactory compromise resulting in a near minimum cost per data mile.

The lifting body vehicle has a length of 21 feet, 7.5 inches. It fits easily under an F-4C vehicle and could be larger both in length and breadth. Like the parawing vehicle, though, the present size is believed to be a satisfactory compromise.

TABLE 5

PARAWING VEHICLE

WEIGHT AND BALANCE

Tail Fins, Nose, and Tail Structure	152 Lb
Avionics and Batteries	84
Control Actuators	23
Recovery System	63
Pressurization and Cooling System	45
Rocket Engine Module	131
Propellant Module (including Tank)	321
Propellants	1816
Liquid Helium	21
Parawing (Maximum Area)	217

	With Large Parawing (Maximum Area)	Without Parawing
Full Weight	2873 Lb	2656 Lb
Full Center of Gravity	178 In. (from nose)	180 In.
Empty Weight	1036 Lb	819 Lb
Empty Center of Gravity	180 In.	182 In.

TABLE 6

LIFTING BODY VEHICLE

WEIGHT AND BALANCE

Airframe	750 Lb
Avionics and Batteries	80
Control Actuators	15
Recovery System	88
Pressurization and Cooling System	38
Rocket Engine Module	131
Turbopump System	55
Propellants	2092
Liquid Helium	13
Full Weight	3262 Lb
Full Center of Gravity	160 In. (from nose)
Empty Weight	1157 Lb
Empty Center of Gravity	166 In.

SECTION 10

OPERATIONS

While it is not possible at this time to obtain a definite ruling, it may be assumed that the previous pattern of allowing unmanned vehicles to fly only over unpopulated or sparsely populated land areas will be followered. There are, however, adequate ranges throughout the world where the vehicle could operate to give world wide turbulence coverage. Figure 52 shows some of the better known unclassified ranges. As the flight program progresses and reliability improves, flights can be made from ranges with less capabilities than the National Ranges in the Continental U.S.A.

10.1 WHITE SANDS MISSILE RANGE

A tentative flight program for 50 flights presented in Table 7 shows initial data gathering flights from White Sands Missile Range (WSMR). Since the initial flights must have more extensive ground support, the capabilities of the WSMR are presented in some detail as representative of those considered necessary. All the ranges shown in Table 7 have sufficient tracking facilities for HI-HICAT operation after the initial data gathering flights have been successfully flown.

White Sands Missile Range covers an area of roughly 40 miles by 100 miles, composing approximately 4000 square miles of the Tularosa Basin. It is flanked on the west by the Organ or San Andres Mountains and on the east by the Sacramento Mountains. Visibility in the area is excellent, being greater than ten miles 96% of the time. The nominal range is 87 nautical miles in a northerly direction. The only exception to this is during 30 select days of the year when the range is extended to 122 nautical miles. This still falls short of the range requirement of the HI-HICAT vehicle.

Further investigation indicated that the Athena launching area at Green River, Utah, would be suitable for the program. Normally this is a 420 nautical mile flight impacting near the RAN site at WSMR. This would give a sufficient range for the proposed HI-HICAT flights. Figure 53 shows the geographical layout of the Green River Launch Site, as presently used for the Athena vehicle. Superimposed is a typical HI-HICAT trajectory.

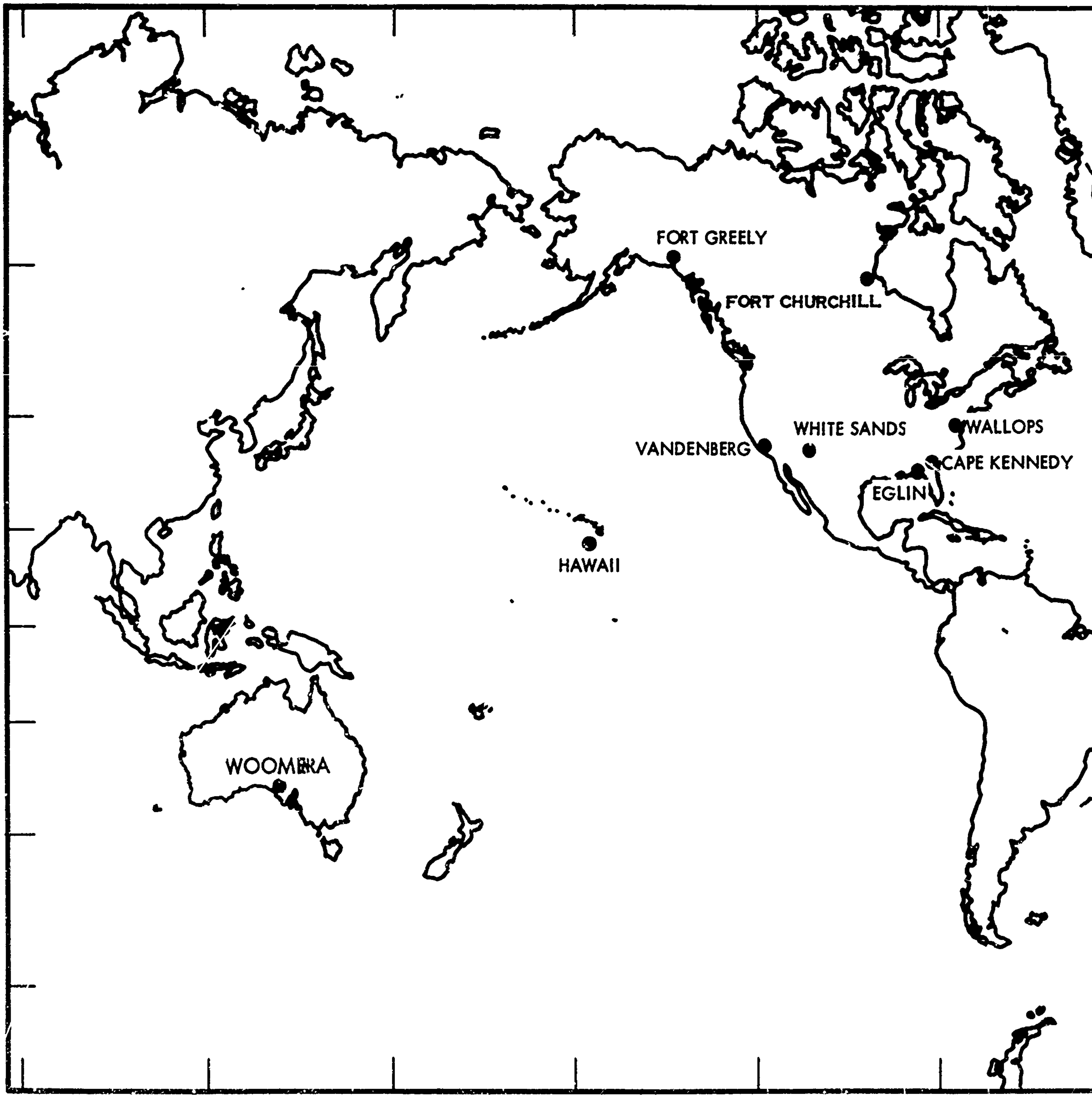
The following organizations and their equipment are available at WSMR to meet the range user's requirements:

A. <u>Measurement Division</u> - Collects all flight data.

B. <u>Data Reduction Division</u> - Reduces data to a form desired by range users.

C. <u>Range Services Division</u> - Provides missile recovery services and operates Army aircraft.

D. <u>Range Instrumentation Development Division</u> - Accomplishes research and development functions associated with


FORT GREELY
FORT CHURCHILL
WHITE SANDS
WALLOPS
VANDENBERG
CAPE KENNEDY
EGLIN
HAWAII
WOOMERA

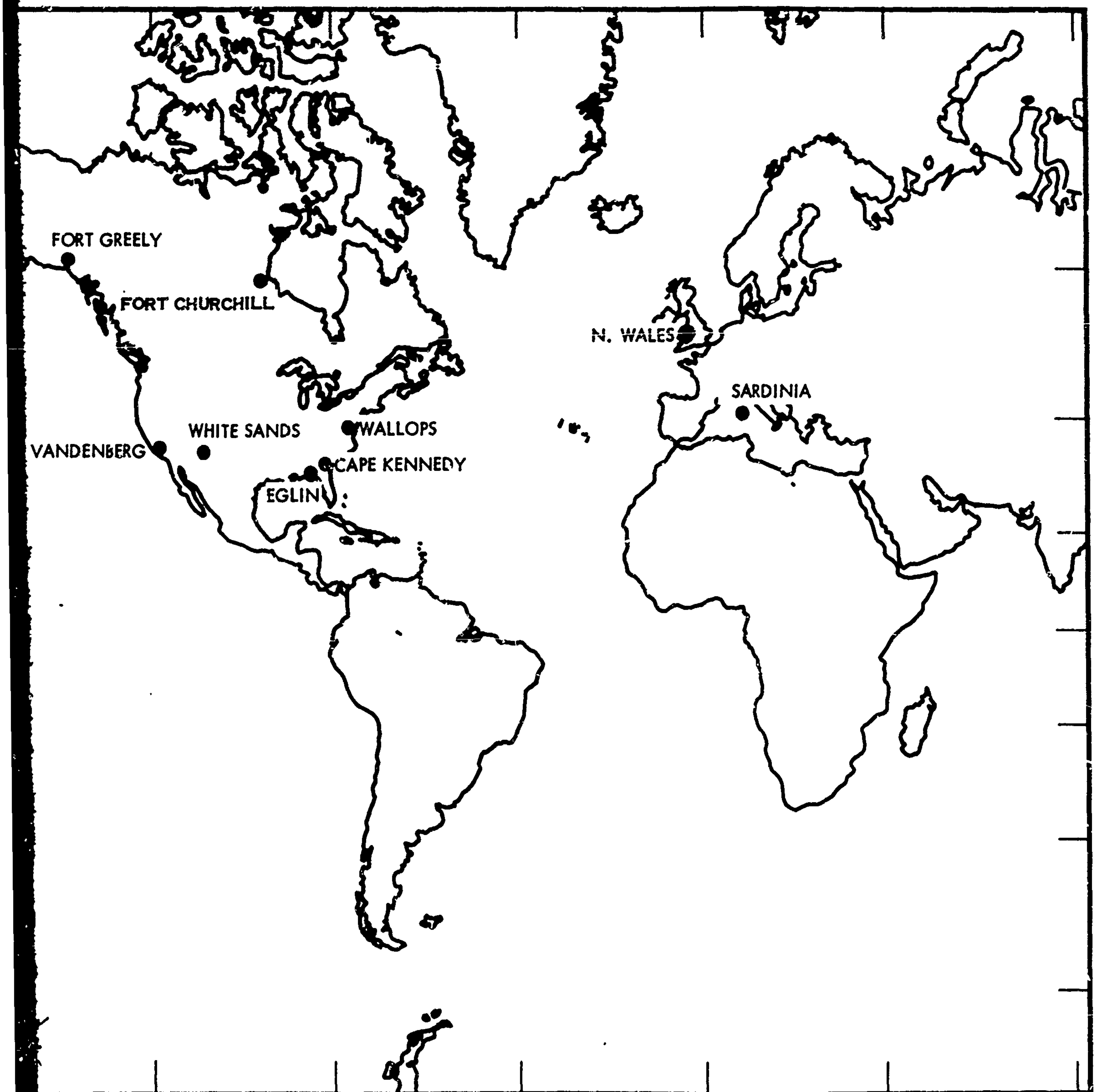


FIGURE 52. POTENTIAL HI-HICAT RANGES

TABLE 7

FLIGHT PROGRAM FOR 50 FLIGHTS

Flights	Location
1 - 5	White Sands Missile Range, New Mexico
6 - 10	Western Test Range, Vandenberg Air Force Base, California
11 - 17	Eglin
18 - 25	Wallops
25 - 35	Fort Churchill, Canada
36 - 40	Hawaii
41 - 50	Woomera, Australia

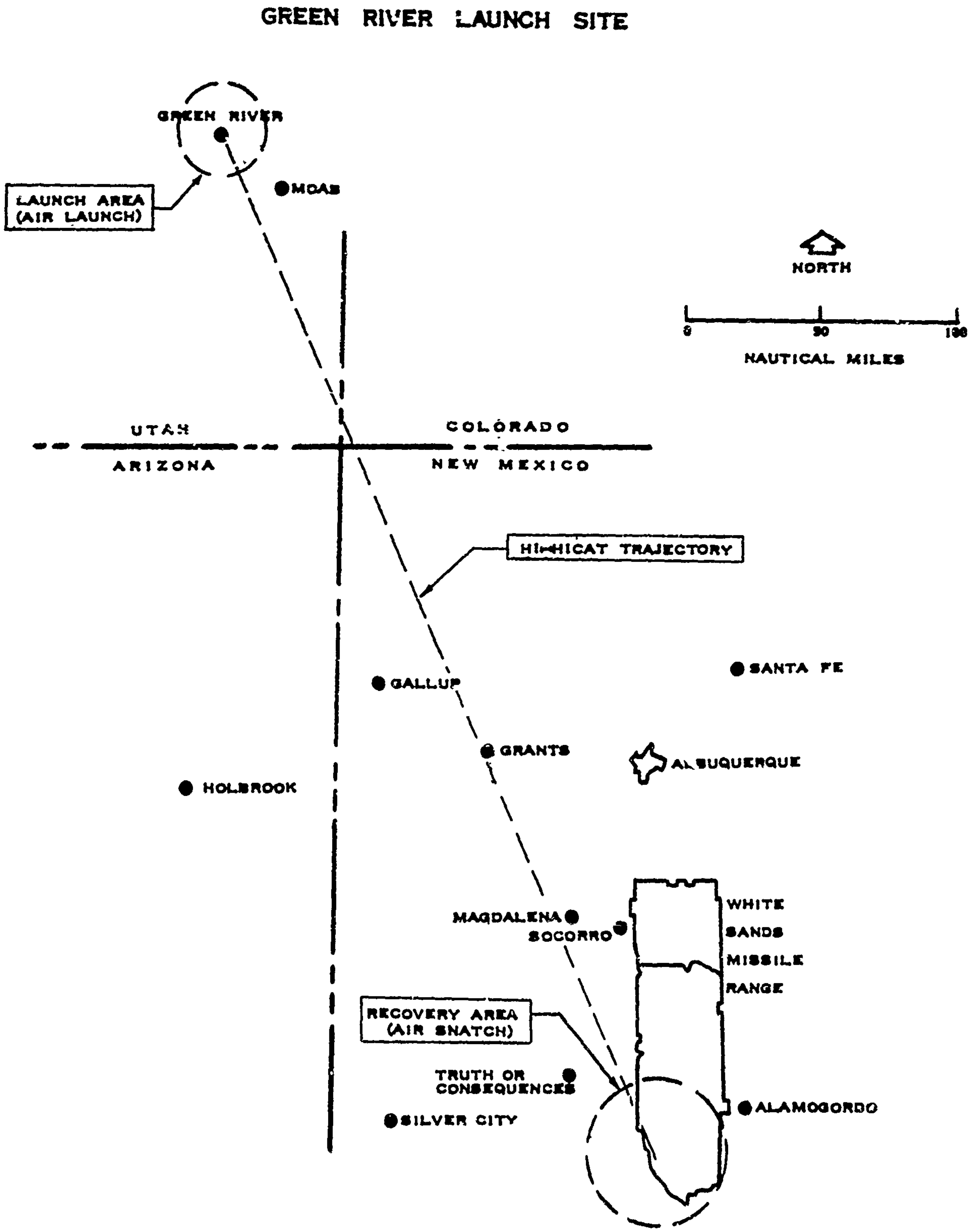


FIGURE 53. GREEN RIVER LAUNCH

the development and/or improvement of data collection/ reduction equipment and techniques.

E. Range Signal Operations Division - Coordinates signal effort in support of range operations.

F. Air Weather (Air Force) - Under coordination control of the Chief, IRM, provides meteorological support to range users.

G. Air Support (Air Force) - Under coordination control of the Chief, IRM, provides aircraft support to range users.

Areas of support available at WSMR that could be used in HI-HICAT flights are given below.

10.1.1 Radar Tracking

Among the tracking equipment available are nine AN/FPS-16 radars. They have tracking rates of 20,000 yd/sec in range, 40 deg/sec in azimuth and 30 deg/sec in elevation, with an RMS range error of 5 yards. Each radar can supply acquisition data, which is referenced to a common coordinate system within WSMR, to all other radars. Radar data are used to provide real-time information to the missile flight safety officer, trajectory data to the project, acquisition data to other range information systems, and vectoring data for drones and target aircraft. Both skin and beacon tracking are employed simultaneously to achieve a high order of reliability.

10.1.2 Optical Tracking

Optical tracking is obtained by use of the Askania high and low speed cameras, at frame rates from 1 to 60 frames per second. The present cinetheodolite system yields position data accurate to approximately ±20 seconds of arc. Derivative velocity and acceleration data are obtained from the position measured.

10.1.3 Telemetry

The range has the capability to receive, decommutate, record, linearize, and scale factor all standard FM/FM, PAM/FM/FM, and PDM/FM/FM telemetry signals transmitted in the 216-to-260-megacycle band. Both crystal-controlled and tunable receivers are used. Magnetic tapes, oscillograph, and pen recorders are used as required. Two 15-foot, parabolic, 18-decibel gain, servo-driven antennas are in use.

10.1.4 Command Control

AN/FRW-2 and AN/URW-15 FM transmitters, operating in the 406-to-549-megacycle band, are used for command control. These transmitters generate, respectively, 500 and 1000 watts of RF power, which is radiated through omnidirectional

antennas. Standard IRIG tone generators are available at all transmitter stations to provide the modulating signals. The tone generators can be controlled remotely from adjacent range control centers. Each transmitter has provisions for external modulation, if required. All stations have two transmitters, with one transmitter serving as automatic standby.

10.1.5 Meteorological Data

Atmospheric data are collected and reduced in support of rocket, missile, and other programs at WSMR. The operational activities center around the collection of data on various atmospheric parameters, prior to, during, and after the launch of a missile. Standard and nonstandard systems and techniques are utilized in obtaining these data. Meteorological data are obtained from observations and measurements of atmospheric pressure, relative humidity, temperature, and wind velocity vectors.

WSMR utilizes a double-theodolite system at selected sites. A theodolite is located at each end of a surveyed baseline. Azimuth and elevation data from each theodolite are recorded when the instruments track a meteorological balloon. A computer automatically reads the theodolite data, performs the required computations, and produces a graph profile of the observed wind components from ground level to 2000 feet. These data are reduced, evaluated, and presented to the user within four minutes from the time the balloon is released.

Six permanent Rawinsonde launch points are established at WSMR. One mobile unit is available to provide off-range or special area support. An instrumentation balloon, which ascends at a rate of approximately 1000 feet per minute to altitudes of between 75,000 and 125,000 feet, continuously telemeters temperatures, relative humidity, and atmospheric pressure data to the Rawin AN/GMD-2 automatic tracking receiver. Azimuth and elevation angles from the tracking unit are recorded at regular intervals.

Meteorological soundings up to altitudes of 600,000 feet can be obtained by use of the Nike Cajun rocket. The Arcas rocket, most frequently fired, can carry a relatively simple telemetry system up to altitudes of 250,000 feet. Parachutes are often launched at apogee. Data resulting from radar track of the parachutes are reduced to wind velocity vectors.

10.1.6 Data Processing

Data processing capabilities are extensive. All standard IRIG telemetry signals can be decommutated, digitized, linearized, scale factored, and stored on either pen recorders, oscillographs or magnetic tape.

10.2 Other Ranges

No other overland range in Continental U.S.A. is adequate for the HI-HICAT mission, but investigation of suitable over-water ranges for early flights was made.

The Pacific Missile Range (PMR) at Point Arguella, California, can supply support matching that of WSMR, and without the tight scheduling problem of the Eastern Test Range (Cape Kennedy). Included in this support is offshore shipboard tracking, and air snatch capability from C130 aircraft based at Edwards Air Force Base. Figure 54 shows the support capabilities of WTR and a possible HI-HICAT trajectory, using their facilities.

WESTERN TEST RANGE LAUNCH

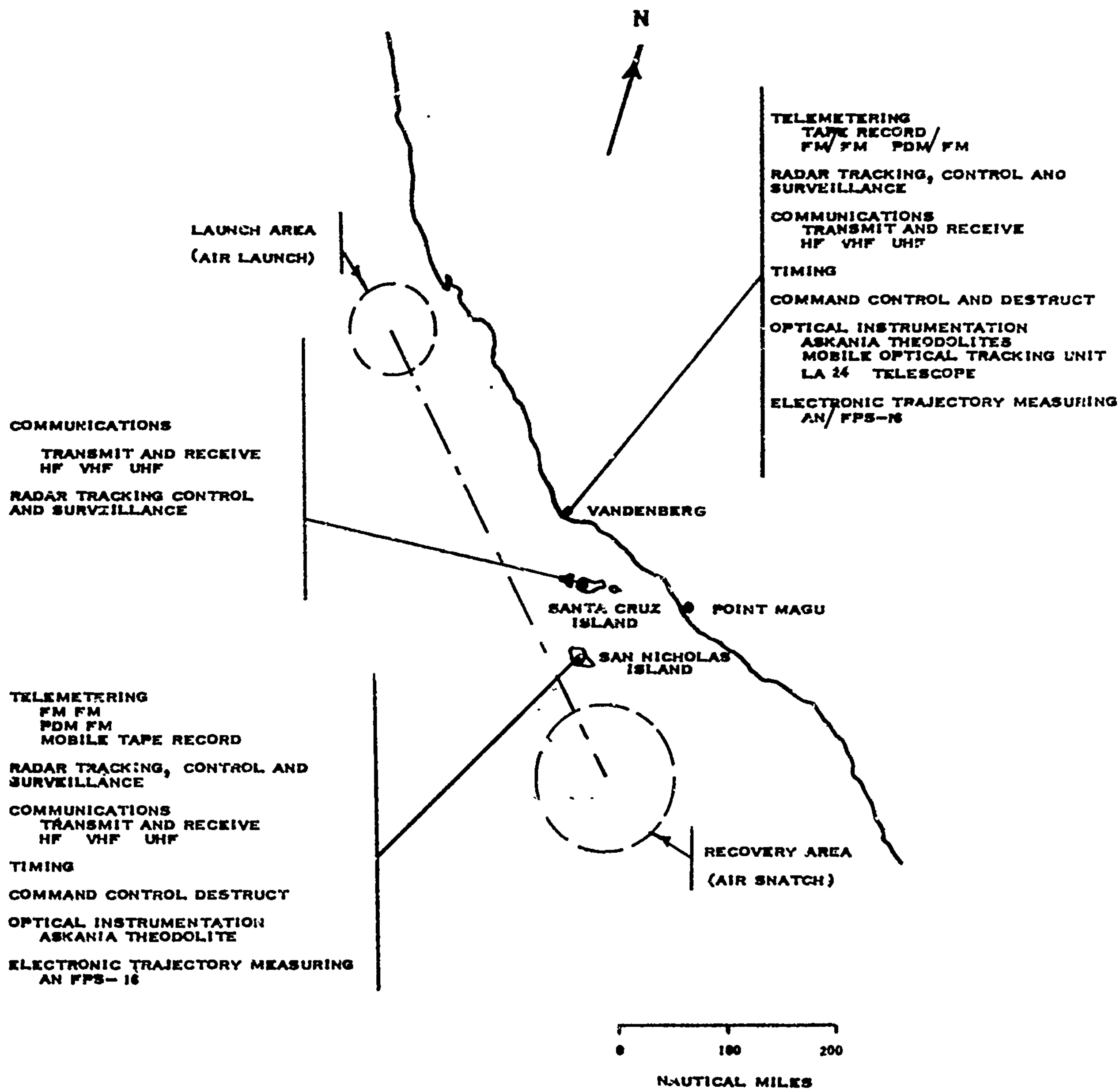


FIGURE 54. WESTERN TEST RANGE LAUNCH

10.3 Summary

To summarize, the recommended mode of operation for the HI-HICAT vehicle is:

1. All flying over non or sparsely populated areas.

2. Maximum use of ground facilities for tracking and data gathering in early flights.

3. Initial data flight from WSMR or WTR.

4. Broadening of operational use of experience and reliability is gained, to eventual world-wide usage with minimum ground support.

SECTION 11

SCHEDULE AND PROGRAM

A program for either a parawing or a lifting body bears such a high degree of similarity that there is no need to differentiate between the two except for a research phase needed for the parawing. The lifting body is a shape which has been studied and tested extensively in recent years for hypersonic vehicles. The data from past programs are readily applicable to the HI-HICAT system. Little interest has existed in parawings for supersonic or hypersonic speeds, primarily because few missions require the extreme altitude capabilities in lifting flight as does the HI-HICAT mission. A parawing research program is outlined which would generate sufficient data to permit establishing a firm foundation for the theory and design of parawings. The program would require slightly over a year and a half, and it must be completed before a parawing vehicle can be developed.

11.1 PARAWING RESEARCH

Little is known concerning the behavior of parawings at supersonic speeds. The National Aeronautics and Space Administration appears to be the only group which has conducted high speed tunnel tests. Only three reports, References 7, 14, and 15, contain usable data. Also, a parawing has not been successfully flown at supersonic speeds. The program given herein is intended to increase the store of high speed aerodynamic and thermodynamic data and permit the development and verification of a flexible wing theory. The outline and schedule for the program is presented in Figure 55. It proceeds by scheduling "hot" and "cold" wind tunnel tests early in the program. These tests are followed by the design and fabrication of experimental gliders. The program ends with the successful flight test of one of the experimental gliders.

11.1.1 Cold Tunnel Tests

High speed tests for aerodynamic force data are planned for wing alone models, a wing plus body model, and a wing plus body plus tail model. The program is based on the utilization of Lockheed's 4 by 4 feet supersonic blowdown wind tunnel located in the Research Center at Rye Canyon near Saugus, California. During a typical blowdown, the angle of attack or the angle of sideslip would be varied slowly while the forces and moments are measured with a six component balance. A number of parawings would be tested (various leading edge sweep angles, canopy curvatures, leading edge radii, leading edge boom curvatures, etc.), at high and low Mach numbers.

Wing deployment tests are also planned in addition to the static force tests described above. Extension would be made at various Mach and Reynolds numbers, and angles of attack and sideslip.

The program calls for the wind tunnel tests to be preceded by aerodynamic and thermodynamic analysis. This analysis is intended to aid in the selection of configurations for testing and to create a theoretical framework by which the test data can be analyzed.

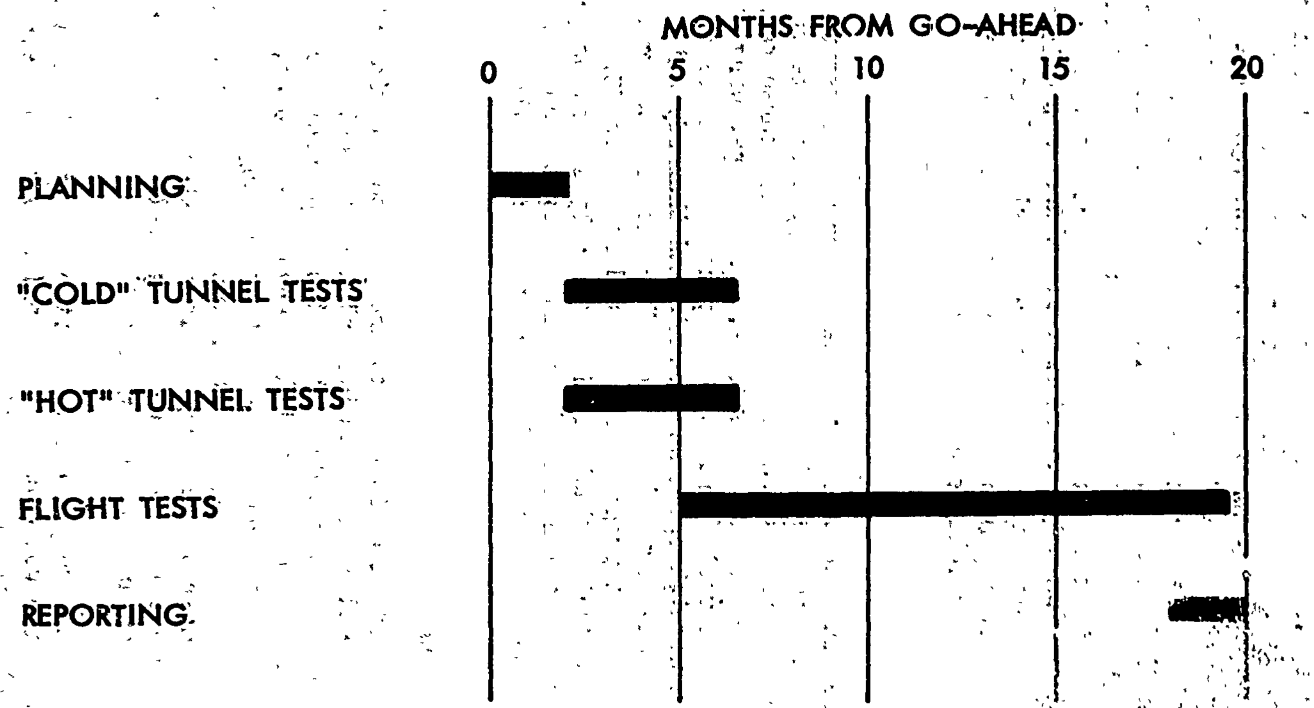


FIGURE 55. SUPERSONIC PARAWING RESEARCH

11.1.2 Hot Tunnel Tests

Hot tunnel tests are required because of the almost complete absence of high temperature data and because of the serious concern expressed about hot spots. At the higher altitudes, considerable cooling is achieved by radiation, but unfortunately the junction between the parawing and the body is a complex thermodynamic configuration having areas with restricted "viewing angles" for radiation. A number of design options are available, such as fairings, heat shield coatings and extremely high temperature materials. Until temperature data are gathered, the shape of the fairings, the thickness of the coatings, and the most suitable material can only be roughly estimated.

The schedule given in Figure 55 shows the hot tunnel tests being conducted concurrently with the cold tunnel tests. The program is based on the utilization of Lockheed's 30 inch diameter hypersonic tunnel.

Rigid models would be constructed which would be instrumented for skin temperatures and pressures. Thermal paint would also be employed. Runs at various angles of attack and sideslip would be completed at several Mach and Reynolds numbers. Tests would be conducted with and without a parawing and with and without wing-body fairings.

11.1.3 Flight Tests

The primary objective of the research program is a successful flight of a parawing vehicle at supersonic speeds; a feat which has not been accomplished to date. The research program described herein calls for a successful flight test of an experimental glider. The test vehicle would weigh roughly 300 pounds and would be boosted from the ground up to the desired altitude and speed by a solid propellant rocket such as the Black Brant IV-A motor. At apogee, the parawing would be deployed and the vehicle would decelerate and glide down to conditions suitable for initiating the parachute recovery package. The only guidance and control function which appears to be essential is a roll attitude control which would position the glider in an upright attitude before the wings are extended.

Three gliders would be fabricated with the expectation that one would successfully complete the test. The choice of three vehicles is in line with the number chosen for similar programs which involve the first time operation of a new system at high speed.

11.1.4 Miscellaneous

A manufacturing technology study would not be necessary. It is expected the need will be eliminated when the manufacturing technology study being conducted by the Space General Corporation of El Monte, California, is completed under contract AF33(657)-10252 (Reference 16). Their investigations are split into three phases of evaluation and design, fabrication and tests leading to the development of structural designs, materials, manufacturing processes and techniques for use in the construction of inflatable re-entry vehicles. The results of their study will be applicable to high speed parawings with rigid leading edges.

11.2 HI-HICAT RESEARCH, DEVELOPMENT, TEST AND ENGINEERING

A schedule for a program leading to the development of a satisfactory HI-HICAT system is presented in Figure 56. It is divided into analysis, subsystem design and tests, instrumentation design, wind tunnel tests, ground tests, flight tests, and miscellaneous. Integrated system analysis is conducted in step with airframe design with due consideration given to the interfaces existing between the various systems. Within the first month the basic vehicle parameters should be established. Also, master scheduling will have formulated the tasks to be accomplished and set up procedures for monitoring the progress of the program.

The propulsion and fuel system, the pressurization and cooling system, the recovery system, the guidance and control equipment, and the instrumentation equipment are items which would normally be subcontracted. The propulsion and fuel system is an integral part of the overall HI-HICAT system and requires a long lead time. Its design is shown as being initiated one month after go-ahead. The design of the other named systems and equipment can be delayed until the integrated system analysis is near completion.

The main pacing item is the first flight which is scheduled for 19 months after go-ahead. The ground tests must, of course, be completed prior to this date, and a one-month lapse in time from the end of ground testing to the start of flight testing is shown. A detailed set of ground tests are specified in line with the requirements for a reliable HI-HICAT system with a high degree of reusability. It appears an integrated series of system tests can be accomplished at the same time that the rocket engine is tested, at a savings in time and effort. These tests will include vibration tests, shock tests, acceleration tests in a centrifuge, mission temperature-pressure simulation, humidity test and antenna pattern tests. One complete vehicle and instrumentation will be scheduled solely for these tests.

The flight tests include 13 buildup flights to maximum speed at an intermediate altitude. All the flights will follow the planned sequence of operation from air launch to air recovery. These tests will be preceded by F-4C launch release test of dummy HI-HICAT vehicles at supersonic speeds.

The next series of flight tests are 5 buildup flights to the maximum design load factor at the maximum speed. These are followed by 5 buildup tests at a high altitude and 4 buildup tests at a low altitude. A total of 8 complete HI-HICAT systems are allotted to the development flight tests. Assuming 14 flights result in total system failure or unsatisfactory data, this yields a flight development program consisting of a total of 13 + 5 + 5 + 4 + 14 = 41 launches.

11.3 HI-HICAT PRODUCTION

The proposed production program, including periods of performance, duration of project by months, and delivery of all items is indicated in Figure 57. The flight tests will be completed in 32-1/2 months after go-ahead, and the research, development, tests and engineering phase will be completed in 33-1/2 months. The production test program will be completed in 4 years after go-ahead for 50 data flights, in 6 years for 500 data flights, and 7-1/2 years for 1000 data flights. This program assumes the use of USAF test facilities and F-4C and C-130 aircraft. Launches will be conducted from only one USAF base or site at any one time in order to reduce HI-HICAT systems and equipment to a minimum.

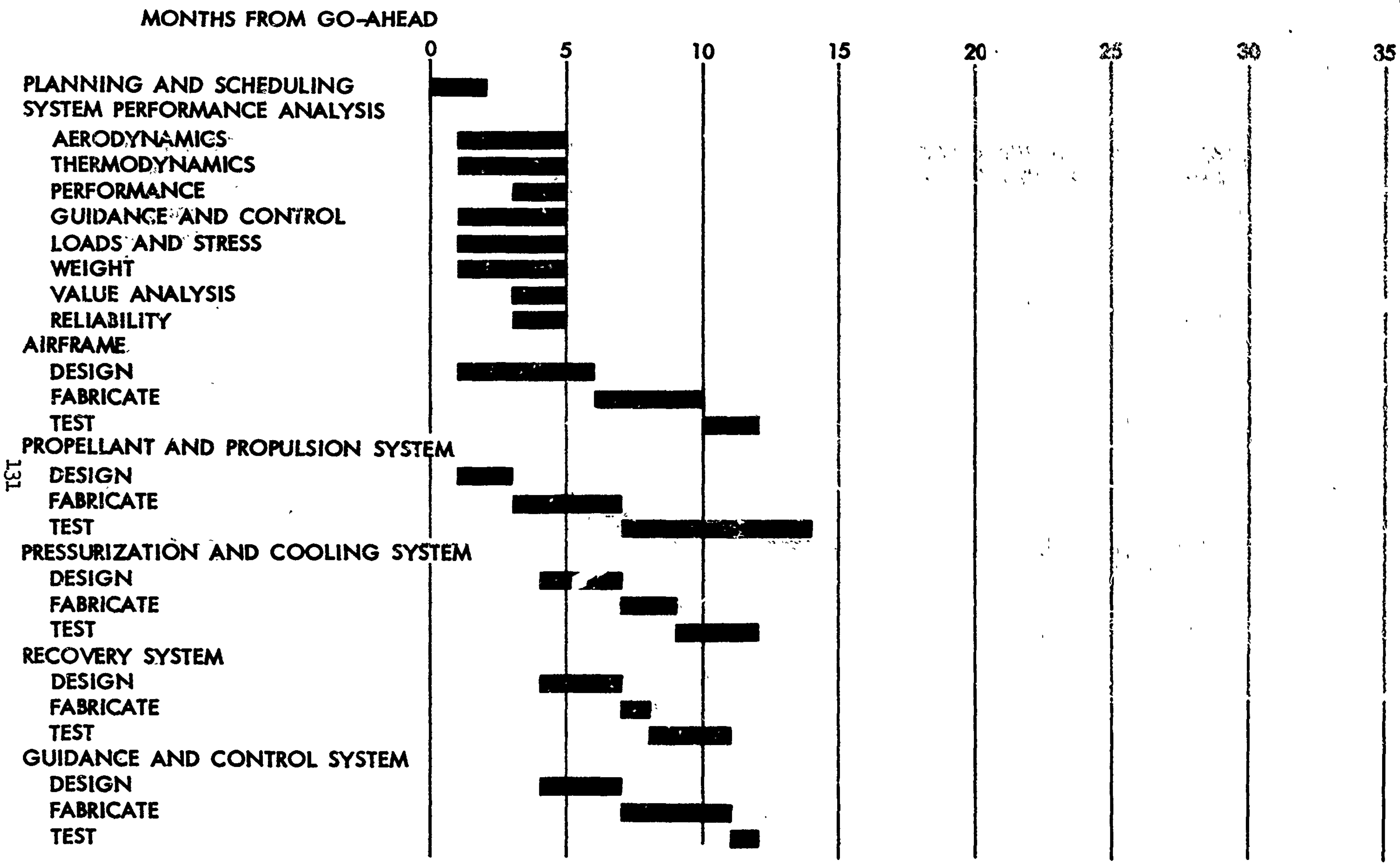


FIGURE 56. DEVELOPMENT, TEST, AND ENGINEERING

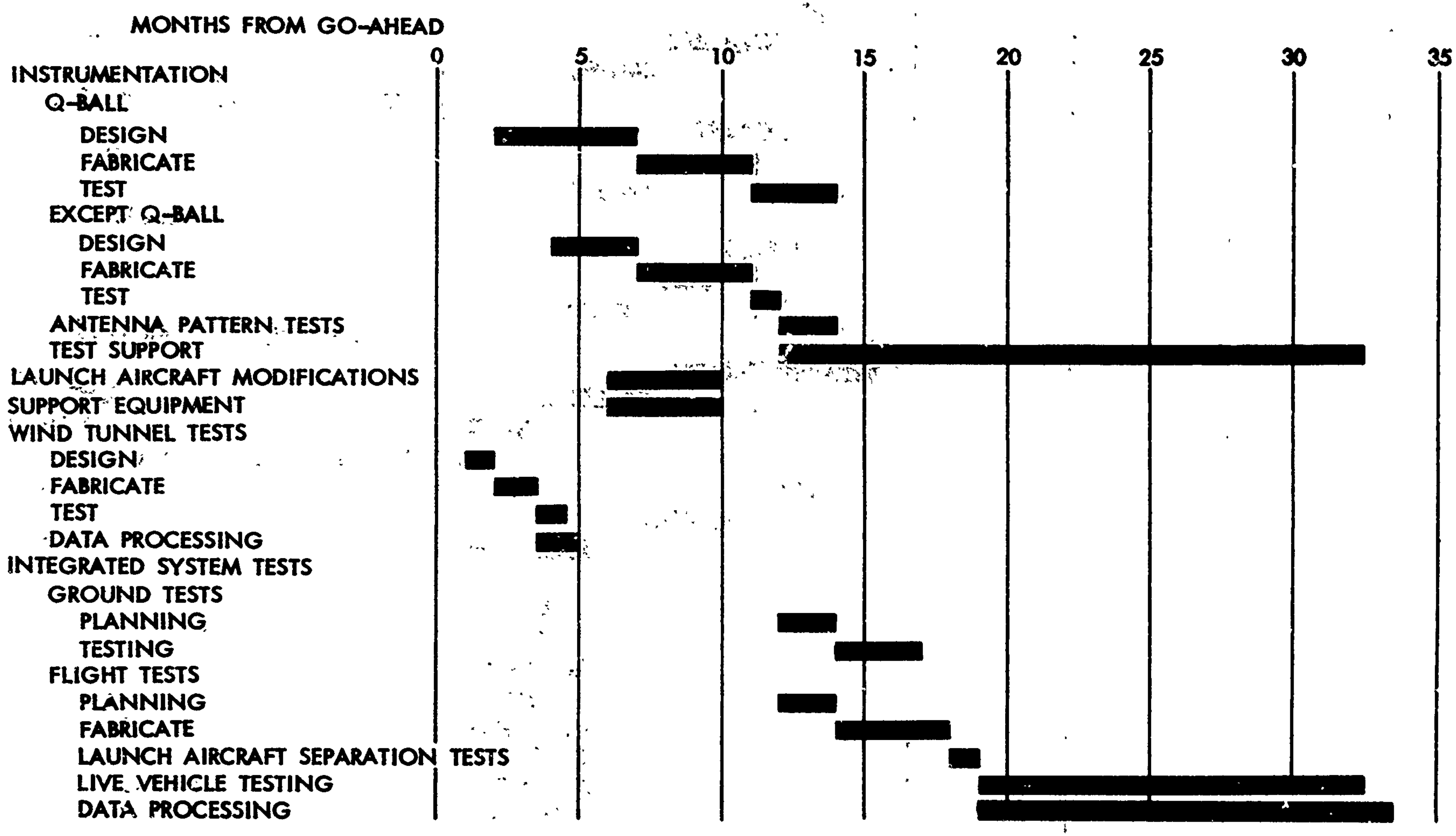


FIGURE 56. (CONTINUED)

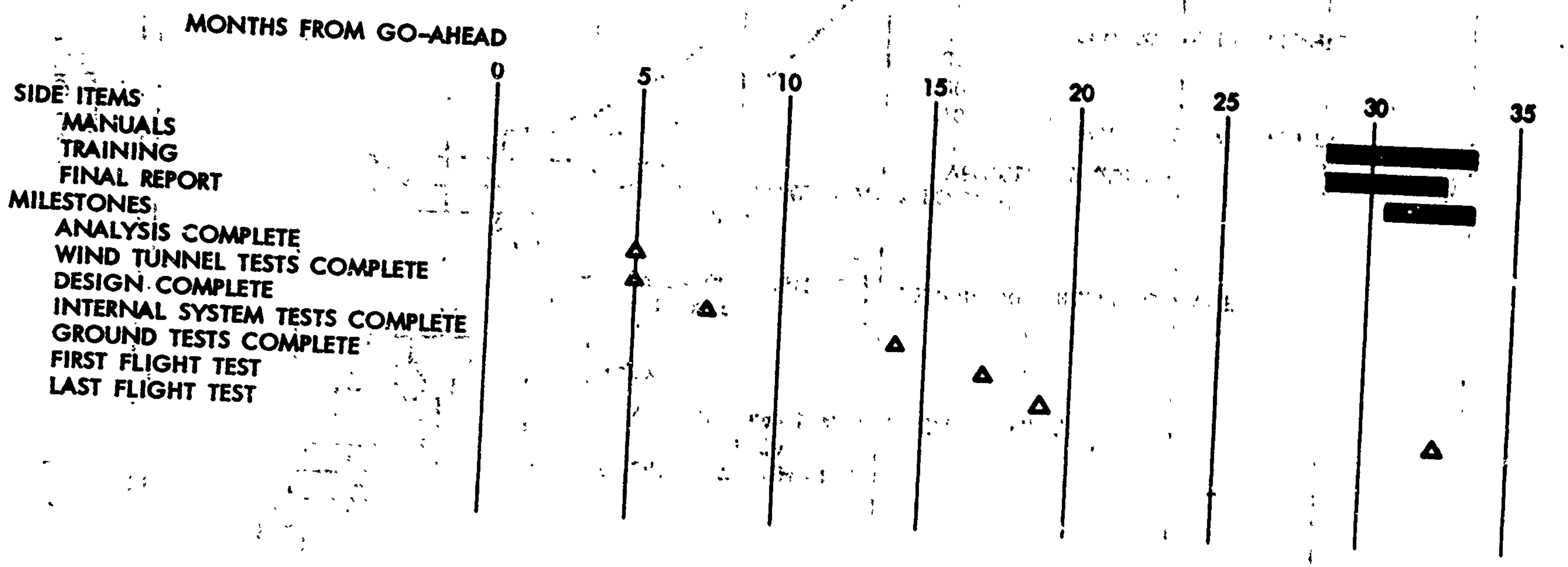


FIGURE 56. (CONCLUDED)

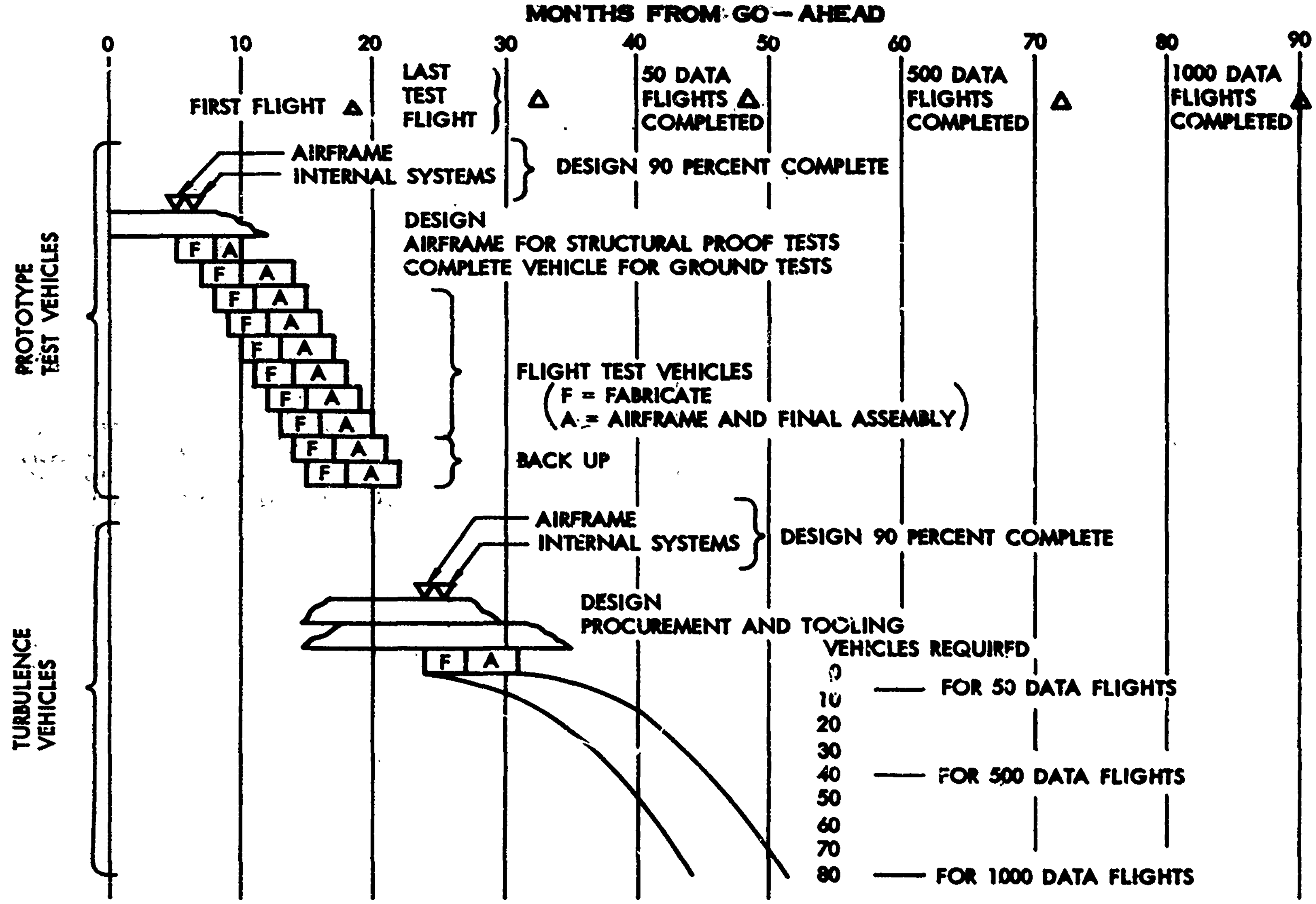


FIGURE 57. PRODUCTION

SECTION 12

COST ANALYSIS

System costs have been developed for a HI-HICAT lifting body vehicle. The costs for developing a HI-HICAT parawing vehicle would be about the same, and hence are not shown. However, a parawing research program must precede a HI-HICAT parawing development program and these additional costs are detailed below. The data for the lifting body vehicle are presented in Table 8. The systems include the RDT&E (Research, Development, Test and Engineering), production and operating phases to successfully complete 50, 500 and 1000 data flights.

The system cost analysis includes all of the items of hardware, facilities and services for the design, procurement and operation of the HI-HICAT system. Inputs to the cost model were derived from several sources including Lockheed historical data, subcontract budget estimates and data from U. S. Government publications. Design requirements indicate that the technology and hardware are within the state-of-the-art. Most of the hardware can be procured from off-the-shelf sources or with a minimum development cost.

All costs are based on 1965 dollars.

12.1 HI-HICAT PARAWING RESEARCH COSTS

As discussed previously, the available information on the behavior of parawings at high speeds is extremely limited. Additional research would therefore be required to bring the parawing up to the existing lifting body state-of-the-art. The cost of this additional work is based on the test program laid out in Section 11 and the schedule of Figure 55.

PARAWING RESEARCH COSTS

(Thousands of Dollars)

Planning	$ 18
"Cold" tunnel tests	180
"Hot" tunnel tests	75
Test vehicles, design to manufacture	1,500
Ground and flight tests	230
Reporting	29
Total	$2,032

12.2 HI-HICAT RESEARCH, DEVELOPMENT, TEST AND ENGINEERING COSTS

The number of vehicles required for the test program was derived in the preceding section. All costs are included to develop and integrate all components, facilities and services necessary to meet the system mission requirements.

TABLE 8

COST ANALYSIS SUMMARY
(THOUSANDS OF DOLLARS)

Number of Successful Data Flights	50	500	1,000
RDT&E Costs (subtotal)	(20,903)	(20,903)	(20,903)
Airframe	8,823	8,823	8,823
Engine	2,400	2,400	2,400
Instrumentation	909	909	909
Subsystems	480	480	480
Other	8,291	8,291	8,291
Production Costs (subtotal)	(3,785)	(18,876)	(35,307)
Flyaway			
Airframe	624	3,202	5,856
Engine	362	2,367	4,708
Instrumentation	892	5,908	11,795
Subsystems	237	1,538	3,050
Other	412	2,299	4,341
Spares	253	1,531	2,975
Other	1,005	2,031	2,582
Operating Costs (subtotal)	(2,460)	(12,819)	(23,731)
Personnel	300	1,500	2,400
Maintenance	767	6,280	12,638
Fuel	87	852	1,705
Other	1,306	4,187	6,988
TOTAL SYSTEM COST	27,148	52,598	79,941
Cost per Data Flight in Dollars	542,960	105,196	79,941

The airframe development cost is derived from Lockheed historical data and similar vehicle studies. Propulsion, pressurization and cooling, recovery, guidance and control, and instrumentation systems costs are based on subcontractor budget estimates.

Other costs include the development of aircraft ground equipment, ground and static testing, test hardware and flight testing.

For this study, it is assumed that the USAF will furnish the facilities and services for flight testing as described in the operations phase, Section 12.4.

It is assumed that the test operating crew will secure the necessary training requirements in the RDT&E phase to efficiently operate the data collection program.

12.3 HI-HICAT PRODUCTION COSTS

Flyaway costs include all material labor and supporting costs for the production of the following number of HI-HICAT vehicles:

50 Data Flights - 6 Vehicles

500 Data Flights - 40 Vehicles

1000 Data Flights - 80 Vehicles

Airframe costs are estimated from Lockheed data.

Engine, instrumentation, and sub-assembly components are taken from subcontractor budget estimates. Installation labor is based on the requirements of the particular component or subsystem.

Due to the small production required, the same tooling is used for the testing and the operational vehicles.

Other costs include technical integration, support equipment, maintenance, sustaining engineering, manuals, and miscellaneous costs.

Initial spares costs are based on 10 percent of the flyaway costs. As the propulsion and recovery systems refurbishment costs are based on the number of vehicle launches and the number of kits required, these costs are not included in the initial spares. These kits are included in the operating refurbishing cost.

12.4 HI-HICAT OPERATING COSTS

For this report certain assumptions are made to arrive at comparative costs of 50, 500 and 1000 data flights.

1. Launches will be conducted from one established USAF base or site at any one time.
2. USAF will provide facilities for storage, maintenance, refurbishing, operating support equipment, and supplies. These facilities costs are not included.

3. Operating range and facilities will be available for data flights as scheduled. Such costs are not included.
4. USAF will maintain all government furnished facilities.
5. The operating team, equipment, and supplies will be transported between locations by the USAF. The cost of operations in foreign areas are not included as the locations are not defined at this time.
6. All launch aircraft (F-4C), and recovery aircraft (C-130) equipped with air snatch equipment and with operating crews, will be furnished by the USAF. These costs are not included herein.
7. Housing and subsistence costs are not included in this report.
8. It is assumed that operating crews will have secured sufficient training during the RDT&E phase that training will not be required for the operating phase of the program.

System operating costs include all personnel, equipment and supplies to accomplish the program for 50, 500 and 1000 data flights.

Personnel includes crews to manage and operate the system. Estimated minimum crew sizes are estimated at 30, 45 and 58 men for 50, 500 and 1000 data flights.

Maintenance and refurbishment costs are based on an initial vehicle turn-around time of 30 days. This performance should improve with experience on extended operating programs. Refurbishing kit costs for the propulsior and recovery systems are based on the number of vehicle launches. Kit costs are based on subcontractors' budget estimates. All other annual replacement costs are based as a percentage of the component or system costs.

Fuel costs are based on current market prices.

Other costs include launch and recovery aircraft operating costs, modification of launch aircraft, data planning, acquisition and analysis.

Launch and recovery aircraft operational costs are computed by extracts from Reference 17. These costs include USAF flight, base, and depot costs based on peacetime aircraft utilization. It is estimated that one F-4C launch aircraft and one C-130 recovery aircraft will service the 50 data flight program. For 500 data flights at 3 flights per week and 1000 data flights at 4 flights per week, two launch and two recovery aircraft will be required.

Aircraft modifications include launch pylons, receiving telemetry and command equipment for the F-4C and the C-130 aircraft.

Data acquisition and analysis costs are based on current HICAT operations. The ground equipment for ground readout and data analysis is assumed to be available and no costs are included for this item.

SECTION 13

ALTERNATE SYSTEMS

In the preceding sections there have been comparisons between the recommended system and various alternatives. The intent and scope of the present contract is the study of a vehicle and instrumentation system which would measure turbulence at extreme altitudes. Considering the difficulties of measuring turbulence at such altitudes, it is appropriate to examine other measuring systems. These alternate systems will not satisfy all the requirements of the present contract, but under some conditions they might produce data with satisfactory accuracy at a lower cost per data bit.

The discussions to follow are brief and are intended solely to acquaint the reader with the possible alternatives. Considerable background information is taken from a study completed for the Aero-Astrodynamics Laboratory, NASA, Marshall Space Flight Center, Huntsville, Alabama, References 18 and 19. That study fulfilled a need for gathering together and analyzing under one cover the many systems for measuring the wind environment of large space vehicles rising through the atmosphere. Here the interest is for horizontal rather than vertical trajectories and for higher altitudes, but the basic measuring principles remain the same.

13.1 REMOTE MEASURING SYSTEMS

Remote systems can be conceived which use the intelligence contained in the laser or radar radiation backscattered by aerosols and inhomogeneities in the atmosphere to deduce the turbulent structure. Such concepts hold little promise because of the extremely poor signal received-to-transmitted power ratios available.

13.2 SOUND SYSTEM

The use of a sound generator was discussed in Reference 19. Unfortunately, a pure tone passing through a turbulent atmosphere is distorted to a significant extent. The small shifts in apparent source frequency due to the winds at altitude would probably be completely masked by the frequency shifts due to the sound waves passing through turbulent layers.

13.3 BALLOONS

Rising or falling balloons follow a trajectory that is usually closer to the horizontal than the vertical. There is no fundamental reason why the data from a Jimsphere or Robin balloon could not be processed to yield the vertical as well as the horizontal winds. The slowly varying component introduced into the apparent winds by the terminal velocity of the balloon must be removed, of course, by numerical filtering or other means. Since an accuracy of one foot per second is desired, only a precision system such as the Jimsphere and FPS-16 radar system, Reference 20, should be investigated. The low cost of this approach is especially appealing.

13.4 SMOKE AND CHAFF TRAILS

Smoke and chaff trails could be laid for a distance equal to the maximum wavelength of interest, that is 75,000 feet, at the apogee of a rocket trajectory. A variation of only ±1000 feet from the desired altitude can be achieved with an apogean velocity of only 1560 fps and a small meteorological rocket could probably be used. Smoke suffers in that good visibility is required. (See Reference 21 for a report on a smoke trail method for vertical wind profiles.) Also, there are problems involved in generating smoke above 60,000 feet as Reference 22 indicates. Chaff appears to be superior as a "tracer" but large Doppler radars must be built or modified from existing equipment.

The possibility of chaff for high resolution wind measurement was studied by the Cornell Aeronautical Laboratory as discussed in Reference 23, Quoting:

"Design concepts were formulated and evaluated for a high resolution (30 m altitude increment) wind measurement concept involving two Doppler radars and a continuous chaff column for rapidly constructing the wind profile from ground to 15 km altitude. The investigation included theoretical interpretation of Doppler spectra, suitable measurement techniques and apparatus for achieving desired sampling capability, and field experiments wherein the general concept was assessed. These simultaneous efforts have led to the following conclusions:

"1. The Doppler radar chaff concept, as devised, is feasible for obtaining wind information of desired accuracy (0.5 - 1.5 m/sec) and spatial resolution (30 m altitude increments). Required equipment to meet these specifications cannot be considered simple or inexpensive; 45 feet to 60 feet diameter antennas, radar transmitters of demanding design, and a unique rocket chaff dispenser are called for. Nevertheless, the envisioned system appears capable of providing, in "real time" and regardless of weather conditions, unique atmospheric wind structure information."

- - - -

"6. Theoretical calculations and chaff tracking experiments have shown that 3 to 5 pounds of chaff (approximately 4 million X-band dipoles per pound) are required per 15 kilometer high column. This concentration is considered suitable in terms of detectable signal return and adequate statistical representation of wind velocities with a sampled volume."

The above quoted accuracy is obtained with a radar sweep over the entire 15 kilometer chaff column in 150 seconds; hence, a number of wind profiles could be generated before the chaff dissipates.

It is recommended further consideration be given to the chaff trails system. It appears a number of "sweeps" could be made of a 75,000 feet trail before the tracer dissipates. This number must be compared to the 25 "sweeps" intended for the HI-HICAT vehicle and the cost per mission and per sweep determined. Although novel, a chaff trail system might generate data at moderate overall costs. The initial costs would be high, however, because of the requirement for a number of precision Doppler radars.

SECTION 14

CONCLUSIONS AND RECOMMENDATIONS

A preliminary design study was conducted for a HI-HICAT vehicle and instrumentation system for operation at altitudes from 70,000 to 200,000 feet which lead to the following conclusions and recommendations:

1. One-stage, unmanned parawing and lifting body designs were evolved for a HI-HICAT vehicle and instrumentation system capable of measuring turbulence at altitudes between 70,000 and 200,000 feet. If the full range of altitudes must be achieved with one system, then a parawing vehicle is optimum. By deploying wings of optimum size, or no wing, the parawing vehicle achieves the greatest range of operating altitudes. The parawing system features a cluster of eight P4-1 rocket chambers, highly pressurized propellant tanks, a cryogenic helium pressurization and cooling system, an inertial navigator, a command control system, and a parachute recovery system designed for an air snatch with a C-130 aircraft. The vehicle is air launched from an F-4C aircraft at speeds near Mach 2. The instrumentation features a one-axis servoed Q-ball, digital data handling equipment, telemetering, and on-board magnetic tape recording.

2. When emphasis is placed on the mid-range of altitudes from 100,000 to 150,000 feet, a lifting body configuration is competitive with and recommended over the parawing. However, a HI-HICAT development program should allow for the exploration of closely related shapes with the goal of achieving substantially better lift-to-drag ratios than those for the vehicle described herein. The internal systems for the lifting body vary little from those for the parawing with the only exception being the use of a turbopump to feed the rocket engine from propellant tanks that are only lightly pressurized.

3. An instrumented YF-12A aircraft is the recommended vehicle for gathering turbulence data at the lower HI-HICAT altitudes. It represents the next logical step up from the present HICAT program with a U-2 aircraft. The cost of obtaining data, although more than that for past turbulence programs with subsonic aircraft, would be considerably less than the cost of the unmanned HI-HICAT system described herein.

SECTION 15

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13. ABSTRACT

A preliminary design study was conducted on an unmanned HI-HICAT (High-High Altitude Critical Atmospheric Turbulence) vehicle and instrumentation system to measure turbulence at altitudes from 70,000 to 200,000 feet. The vehicle configuration selected as optimum for this extreme range of altitudes is a parawing. For the study, emphasis was placed on designing a system for the middle portion of the altitude band from 100,000 to 150,000 feet. In this band a lifting body configuration is competitive with the parawing. Both systems feature a one-stage vehicle which is air launched from an F-4C aircraft at supersonic speeds. A cluster of eight P4-1 rocket chambers accelerates the vehicle up to cruise speed. The vehicle cruises in horizontal flight at speeds as high as Mach 6 until propellant exhaustion or until the sustainer engine is shut down. It then decelerates at the cruise altitude to obtain additional data miles. Recovery is initiated when the vehicle slows down to Mach 1.5. An air snatch completes the mission. Turbulence data is gathered by a digital system and stored on a magnetic tape recorder and telemetered back to the launch aircraft, the recovery aircraft, and any available ground station. An inertial navigator supplies attitude angle and acceleration data, but the fine scale attitude motions in turbulence are obtained from a package of three precision rate gyros. A one-axis, servoed Q-ball is recommended as the flow direction sensor. The total RDT&E, production, and operating cost of 500 data gathering flights is estimated at $53 million.

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Air Launched Drone						
Aerial Retrieval						
Rocket Propelled Drone						
Parawing						
Lifting Body						
Inertial Guidance						
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